AFFTC-TR-98-05

CHANGE IN RANGE FACTOR AS A RESULT OF AN APPLICATION OF AN AVIATION POLISH TO A T-38A AIRCRAFT (HAVE SLICKER)

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NOVEMBER 1998

FINAL REPORT

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AIR FORCE FLIGHT TEST CENTER
EDWARDS AIR FORCE BASE, CALIFORNIA
AIR FORCE MATERIEL COMMAND
UNITED STATES AIR FORCE

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This report presents the results of an evaluation of a change in range factor as a result of an application of Racer's Edge polish to a T-38A aircraft (HAVE SLICKER). The objective of this effort was to characterize potential range factor (true airspeed multiplied by aircraft gross weight divided by fuel flow) changes as a result of the polish application. Testing was conducted by the USAF Test Pilot School Class 97B at Edwards AFB, California, from 16 March to 9 April 1998. Testing was requested by the Air Force Flight Test Center Single Face to Customer Office, at Edwards AFB, and was conducted under a Cooperative Research and Development Agreement number CR980100.

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PREFACE

This report presents the results of an evaluation of a change in range factor as a result of an application of Racer's Edge polish to a T-38A aircraft (HAVE SLICKER). The objective was to characterize range factor (true airspeed multiplied by aircraft gross weight divided by fuel flow) changes due to polish application on a Northrop T-38A aircraft. Testing was conducted at Edwards AFB, California, by the USAF Test Pilot School from 16 March through 9 April 1998. Testing was requested by the Air Force Flight Test Center Single Face to the Customer Office at Edwards AFB and conducted under Cooperative Research and Development Agreement number CR980100.

Sincere appreciation is expressed to Mr. Larry Sweetser of American Aviation & Toolcraft for loaning the test team a Taylor-Hobson Surtronic 10 Ra surface analyzer. Special thanks are extended to Messrs. Pete Jozsa, T-38 Shadow Maintenance (412 TS/LGFSG), and Dick Shutte, Special Instrumentation (412 LG/LGMSS) for outstanding support and troubleshooting throughout the HAVE SLICKER test.

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EXECUTIVE SUMMARY

This report presents the results of an evaluation of a change in range factor as a result of an application of Racer's Edge polish to a T-38A aircraft (HAVE SLICKER). The HAVE SLICKER test program was a limited evaluation of an aviation polish developed and manufactured by Racer's Edge and marketed by American Aviation & Toolcraft under a USAF Cooperative Research and Development Agreement (CR980100) between the Air Force Flight Test Center (AFFTC), Edwards AFB, California, and American Aviation & Toolcraft, Quartz Hill, California. The testing was conducted by the USAF Test Pilot School at the AFFTC. Fourteen Northrop T-38A (USAF S/N 68-8153) test sorties, totaling 19.4 flight test hours were conducted from 16 March through 9 April 1998.

The test item, Racer's Edge Polymer Aviation Polish, was a water-soluble, thin-film polish. Application of this polish deposited a limited buildup (1 micron, or 0.00005 inch) on the aircraft surface. This buildup was considered a minor modification and did not require removal after testing.

The overall test objective was to characterize the average range factor (true airspeed multiplied by aircraft gross weight divided by fuel flow) change resulting from Racer's Edge polish application to the T-38A aircraft. The specific test objective was to determine, within a 90-percent confidence level, if there was a range factor change of at least 1.2 percent attributable to the application of Racer's Edge polish. The estimated instrumentation resolution was ±0.6 percent; therefore, a 1.2 percent change in range factor was the estimated minimum resolvable difference between a baseline and treated data point. The measure of performance was a comparison between the range factor of the baseline (untreated) aircraft versus the treated test aircraft using the weight to ambient air pressure ratio flight test technique.

The overall HAVE SLICKER test objective was met. While flight test results revealed a trend toward improved average range factors at both altitudes, the improvement was less than the ±0.6 percent instrumentation uncertainty. Test results revealed a 0.7 percent improvement in the range factor at 11,000 feet pressure altitude and a 0.3 percent improvement at 13,500 feet pressure altitude; therefore, a 90-percent confidence in a measurable improvement in the range factor was not attained at either altitude, and no meaningful statistical confidence could be obtained.

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INTRODUCTION

GENERAL

This report presents the results of an evaluation of a change in range factor as a result of an application of Racer's Edge polish to a T-38A aircraft (HAVE SLICKER). The overall test objective was to characterize the range factor (true airspeed multiplied by aircraft gross weight divided by fuel flow) change as a result of the Racer's Edge polish application to a Northrop T-38A aircraft.

Testing was conducted by the Air Force Flight Test Center (AFFTC) USAF Test Pilot School (TPS), Edwards AFB, California, under USAF Cooperative Research and Development Agreement number CR980100, between the AFFTC and American Aviation & Toolcraft, the marketing agent for Racer's Edge aviation polish. The test team consisted of three test pilots, one flight test navigator, and two flight test engineers. A total of 19.4 hours of flight tests were conducted using a single T-38A aircraft, USAF S/N 68-8153, from 16 March through 9 April 1998. Seven baseline (without polish application) test sorties and seven treated (after polish application) test sorties were flown and the results directly compared. Flight tests were conducted in the Edwards AFB R-2508 complex and the airspace between Daggett and Needles, California.

A similar test project, HAVE SLICK (Reference 1), was conducted by the USAF TPS in 1995. The HAVE SLICK test team tested the Thin Film Systems' MICROCLEAN™ product on a T-38A aircraft for drag reduction effects. Results indicated a 1-percent improvement in specific range (range factor divided by aircraft gross weight) and drag reduction as a result of the chemical treatment.

PROGRAM CHRONOLOGY

The HAVE SLICKER test project was conducted from 7 January through 9 June 1998. Flight tests were conducted from 16 March through 9 April 1998. Significant project milestones are summarized in Table 1.

TEST DESCRIPTION

Test Item Description:

Racer's Edge was a water-soluble polish. Application of the polish deposited a limited buildup (1 micron, 0.00005 inch) on the aircraft surface. The polish was nontoxic, biodegradable, and required no HAZMAT devices, clothing, or protocol throughout the test. Racer's Edge polish could be applied to painted and clear-coated metal or composite surfaces. The polish dissipates in 6 to 12 months and did not require removal. Further polish information and the material safety data sheet can be found in the Program Information Document (Appendix A).

Test Aircraft Description:

The test aircraft, a Northrop T-38A, was a two-place, twin-turbojet, supersonic trainer. The fuselage

Table 1
PROGRAM SCHEDULE

Date	Event
7 Jan 98	Program Introduction
12 Feb 98	Test Plan Working Group Meeting
17 Feb 98	Test Plan Complete
2 Mar 98	Technical and Safety Review Board
16 Mar 98	Flight Tests Begin
9 Apr 98	Flight Tests Conclude
17 Apr 98	Draft Technical Report Complete
1 Jun 98	Final Technical Report Complete
8 Jun 98	Final Oral Report

had an area-rule shape, with moderately swept-back wings and empennage. The aircraft had irreversible flight control systems. It was powered by two J85-GE-5 engines, each producing 2,050 pounds of thrust in military power (sea level, static).

The production noseboom of the test aircraft was replaced with a flight test noseboom. In addition, the aircraft was also equipped with sensitive instruments to gather performance data. The aircraft was considered production representative since the modifications had negligible effects on the aircraft's cg[1] and aerodynamics, and no effect on the aircraft's thrust characteristics. A complete description of the T-38A and class II modifications are contained in the T-38A Flight Manual (Reference 2) and the aircraft Modification Flight Manual (Reference 3). The polish was applied to approximately 88 percent of the aircraft's wetted area in the cruise configuration. Application of the polish was covered by a modification note.

Test Instrumentation Description:

The T-38A test aircraft was equipped with a Rosemount Model 850A yaw angle-of-attack (AOA) and Pitot-static noseboom, which provided angle-of-sideslip, AOA, and Pitot-static sensor information. A Rosemount Model E102AL total air temperature probe was also incorporated on the lower portion of the forward fuselage to provide total air temperature data. Sensitive airspeed indicators, altimeters, and Mach meters replaced production versions in both cockpits. A complete description of all modifications can be found in Reference 3. In addition, the aircraft was equipped with a Metraplex Data Acquisition System (DAS) containing a ¼-inch tape recorder for collecting aircraft test parameters. A detailed description of the Metraplex DAS can be found in the USAF TPS Instrumentation Handbook (Reference 4). Calibration records for all instrumentation and cockpit instruments were maintained by the USAF TPS Technical Support Division. Specific calibration records, resolutions, and accuracies can be found in Appendix B.

Data Acquisition System:

The T-38A test aircraft's onboard DAS recorded performance data. The onboard DAS was critical for accurate data acquisition and was the main go/no-go criteria. The DAS data were processed using the existing USAF TPS Aydin processor. The T-38A DAS collected airspeed, altitude, fuel flow, vertical velocity, AOA, fuel used, total air temperatures, and horizontal stabilator position data during each test point. A complete listing of DAS recorded parameters is presented in Appendix B.

All DAS data were backed up by hand-held data collected at regular intervals during the 3-minute test points. Hand-held data included fuel counter, fuel quantity, altitude, indicated velocity, vertical velocity, fuel flow, engine rpm, and qualitative pilot comments to include turbulence. These comments were used to assess the overall test point quality. Postflight data reduction was conducted to verify data quality and to calculate the aircraft's range factor for each test point. The DAS file was downloaded into the USAF TPS Aydin processor for conversion to engineering units and then converted to a USAF TPS flight test analysis system DAS file. This file was subsequently converted to an ASCII file which was used in the test-team developed data reduction spreadsheet. Appendix C contains the equations used to calculate range factor.

TEST OBJECTIVE

Overall Test Objective - Range Factor Characterization:

The overall test objective was to characterize the T-38A aircraft range factor change as a result of application of the Racer's Edge polish. Specifically, the range factor of the T-38A test aircraft with and without application of the polish was compared, using the gross weight to ambient air pressure ratio (W/δ) flight test technique (FTT).

Specific Test Objective - Range Factor Comparison:

The specific objective was to determine, within a 90-percent confidence interval, if a change in range factor of at least 1.2 percent occurred as a result of an application of the Racer's Edge Polymer Aviation Polish on the T-38A test aircraft. The 1.2-percent change was the estimated uncertainty in range factor due to instrumentation uncertainty.

[1] All cg's in this report are longitudinal cg's, in percent mean aerodynamic chord.

Measure of Performance - Range Factor Comparison:

Based on previous testing (HAVE SLICK, Reference 1), the test team estimated 25 test points at each W/δ and Mach number combination were required to compare the range factors of the baseline versus the treated test aircraft. Appendix D shows specific W/δ test points.

LIMITATIONS

A limitation governing the conduct of this flight test was instrumentation uncertainty. The resolution and uncertainty of the T-38A DAS limited the FTTs available. The W/δ FTT was the best technique available to obtain repeatable cruise data within the T-38A DAS limitations. Based on the ±0.6-percent instrumentation uncertainty in range factor, the polish needed to improve the range factor by at least 1.2 percent in order to conclude that a change in range factor was evident. This instrumentation uncertainty was one of the primary factors in test point selection (Appendix C).

TEST AND EVALUATION

GENERAL

The overall test objective was to characterize the range factor change as a result of an application of the Racer's Edge polish to a T-38A aircraft. If Racer's Edge polish was proven to improve range factor, there could be potential savings in fuel cost for both military and commercial flight operations. However, the test had to be performed to a high degree of certainty since the potential improvement due to the polish application was predicted to be the same order of magnitude as the instrumentation uncertainty.

TEST AIRCRAFT CONFIGURATION CONTROL

All tests were conducted in the cruise configuration (flaps, speedbrakes and landing gear retracted). The evaluation used the W/δ FTT as described in the USAF TPS Performance Phase Planning Guide (Reference 5). Test aircraft cg was maintained at 18 ±0.1 percent during flight testing (Appendix F). The test aircraft was hangared between test flights to minimize environmental effects.

TEST PREPARATION

Test Item Application:

The test aircraft was washed using standard maintenance procedures prior to the first baseline test sortie and again prior to application of the polish. Two T. Brennan, Inc. (subcontractors to American Aviation & Toolcraft) personnel applied the polish to the test aircraft. Application of the polish was limited to exposed, external, painted surfaces in the cruise configuration. No polish was applied to the noseboom, Pitot-static ports, AOA vanes, total air temperature probes, engine nozzles, inside of the engine inlets, canopy, black anti-glare surfaces, landing gear (other than the exterior surface of the gear doors), or antennae. The total area that remained free of polish was approximately 115 square feet, or 12 percent of the total aircraft wetted area in the cruise configuration. No polish was applied between treated test flights.

Test Aircraft Weight and Balance:

The aircraft weight and cg, as well as fuel quantity indications, were measured with various fuel loads (900, 2,200 and 3,500 pounds fuel) at two pitch attitudes, 0- and 5-degrees nose high. These pitch attitudes bracketed the predicted AOA at the various test points. A gross weight and longitudinal cg table was developed to record measured variations in fuel quantity indications and cg locations as a result of aircraft pitch attitude changes, fuel burn, and fuel imbalances (Tables F1 and F2).

To further minimize gross weight and cg variations, the test team flew a constant pairing of pilot/engineer aircrew. Each specific crew had specific Weight and Balance Clearances (Forms F) (Appendix F) generated to reflect each crew's takeoff gross weight and cg. Based on fuel tank moment data from Technical Order (T.O.) 1T-38A-5 (Reference 6) and the weight & balance measurements, a crew-specific fuel burn curve (Figure F1) and table (Table F3) was created to reflect the desired fuel distribution. This table was used to keep the cg constant at 18 ±0.1 percent while cruise data were collected.

TEST PROCEDURES

Aircraft Fueling Procedures:

An accurate fuel weight was required to achieve a precise starting gross weight for each test sortie. The aircraft production fuel gauges were calibrated on the weight and balance stand and found to be accurate to 30 ±9 pounds (the weight and balance stand's uncertainty was ±0.07 percent. This value multiplied by the maximum aircraft gross weight, 12,973 pounds, resulted in the stand's uncertainty of ±9 pounds). Fuel was loaded at 25 pounds per square inch line pressure for each sortie and the single point refueling automatic shutoff was used to stop the fuel flow. Using this procedure, accurate and repeatable starting fuel weights were obtained.

Following each test sortie and just prior to refueling, maintenance personnel collected a 1-quart sample of JP-8 fuel from the aircraft's forward fuel

dump valve. Phillips Laboratory, Edwards AFB, California, analyzed the sample for energy content and fuel density at representative test point and refueling temperatures (Appendix G). An additional 1-quart sample was taken from the refueling truck line to measure the temperature of the onloaded fuel.

Surface Roughness Measurement:

Surface roughness measurements were taken with a Taylor-Hobson Surtronic 10 Ra surface analyzer supplied by the customer, American Aviation & Toolcraft. The surface roughness measurements were taken on three occasions: prior to the first baseline test sortie, after aircraft polish application, and after the last treated test sortie. The first two measurements were taken to compare surface roughness before and after polish application, and to determine test altitude and Mach number as discussed in Appendix E. The last measurement was taken to determine changes in surface roughness during the treated test sorties. For all measurements, 40 locations were randomly selected around the airframe and averaged. The results of the surface roughness measurement are shown in Table 2. The results revealed the aircraft surface to be smoother after polish application. No polish was applied between treated test flights. Measurements taken after the last test sortie revealed no degradation in skin smoothness.

Engine Parameters:

Since any degradation in engine performance could severely impact the outcome of the test results, engine performance was monitored throughout the entire flight test phase. For each flight, a 2-minute engine run at military thrust was accomplished to collect engine data which included engine rpm, fuel flow, exhaust gas temperature, nozzle position, ambient air temperature, and pressure altitude.[2]

The engine runs showed no significant changes in engine performance during the course of the test. The results are presented in Table 3.

Flight Test Technique:

All test sorties were flown using the W/δ FTT which required the aircraft to be flown at a given Mach number and W/δ. As the aircraft gross weight decreased from fuel burn, the pressure altitude was increased to maintain a constant W/δ. This FTT is described in Reference 5. Each test point was flown for a minimum of 3 minutes to ensure an accurate average fuel flow. The W/δ FTT reduced the number of gross weight and altitude conditions that would otherwise be required. Data reduction required the W/δ to be maintained to within ±2 percent. A more detailed description of this FTT is described in the USAF TPS Performance Phase Planning Guide (Reference 5). Specific data bands and tolerances are summarized in Table 4.

Fuel Heating Value Ratio:

The heat of combustion for each sortie's specific fuel sample was determined by Phillips Laboratory by an average of three bomb calorimeter tests. These values were then divided by the average heat of

Table 2
SURFACE ROUGHNESS MEASUREMENTS

- Measurements made with Taylor-Hobson Surtronic 10 Ra
- S/N 1344647
- Calibrated 20 October 1997
- Resolution = ±4μ inch[1], Uncertainty = ±5 percent or reading or ±1μ inch[1], whichever is greater

	Low Value (mil)[2]	Average Measurement (mil)[2]	High Value (mil)[2]
Prior to First Baseline Test Sortie	0.24	0.26	0.27
After Polish Application	0.11	0.19	0.27
After Last Treated Test Sortie	0.10	0.16	0.25

Notes: 1. μ inch = 10^{-6} inches
2. 1 mil = 10^{-3} inches, or 0.001 inches

[2] All altitudes in this report are pressure altitude in feet, unless otherwise specified.

Table 3
ENGINE RUN ANALYSIS

- T-38A, S/N 68-8153
- J85-GE-5 engines
- JP-8 fuel
- Military power
- Ground level, static

			Engine RPM (pct)		EGT (deg C)		Nozzle Position (pct)		Fuel Flow (lb/hr)	
	OAT	PA	Left	Right	Left	Right	Left	Right	Left	Right
First Baseline Sortie	51	2,192	100.7	100.0	640	640	12	20	2,150	2,000
Last Test Sortie	50	2,211	100.7	100.4	640	635	13	20	2,200	2,000

Notes: 1. OAT - airfield ambient temperature, in degrees Fahrenheit, as reported by the base weather shop
2. PA - airfield pressure altitude, in feet, as reported by the base weather shop
3. EGT - exhaust gas temperature

Table 4
W/δ FLIGHT TEST TECHNIQUE DATA BANDS AND TOLERANCES

Parameter	Data Band	Tolerances
Pressure Altitude, ft	±100	±100
Vertical Velocity, ft/min	--	±100
Indicated Airspeed, KIAS	±2	±2
W/δ, pounds	17,600	±2 percent
cg, percent MAC	18	±0.1

Notes: 1. '---' – not applicable
2. W/δ – aircraft gross weight
3. MAC – mean aerodynamic chord

The average heat of combustion was 18,636 British thermal units per pound of fuel. The fuel heating value ratios used throughout the flight tests are presented in Appendix G.

Fuel Density:

The fuel density for each sortie's specific fuel sample was determined by Phillips Laboratory at three different temperatures: the highest and lowest DAS measured test point fuel temperatures, and the lowest of the measured aircraft fuel or refueling truck fuel temperatures. A linear correlation was used to model the density-temperature relationship, which is presented in Appendix G. There was no significant variance in densities from different fuel lots taken at the same temperature, so a uniform lot assumption was made. The density at each test point was determined at the average DAS-measured fuel temperature for the point using the fuel density model. The average DAS-measured volumetric fuel flow was multiplied by this density value to find the average mass fuel flow for each point.

TEST RESULTS

Baseline Range Factor Characterization:

A total of seven baseline test sorties totaling 8.0 flight hours were flown. The test sorties were flown at 0.77 Mach number and a W/δ ratio of 17,600 pounds. Data points flown below 12,000 feet pressure altitude were standardized to 11,000 feet, while those at 12,000 and above were standardized to 13,500 feet. A total of 62 baseline data points (21 data points standardized to 11,000 feet and 41 data points standardized to 13,500 feet) were flown. Of the 62 data points collected, 42 points were considered good quality; the other 20 test points were of marginal or poor quality due to

turbulence or flying outside parameter tolerances. Within a 90-percent confidence interval, the average range factor for 11,000 feet was 2,055 ±6 nautical air miles (NAM). For 13,500 feet, the average range factor was 2,020 ±5 NAM. Baseline testing was terminated when additional test points did not significantly change the range factor values or confidence intervals.

Treated Range Factor Characterization:

A total of 7 treated test sorties totaling 7.0 flight hours and a total of 57 treated data points (29 data points standardized to 11,000 feet and 28 data points standardized to 13,500 feet) were flown. Out of the 57 data points collected, 45 points were considered good quality; the other 12 test points were of marginal or poor quality due to turbulence or flying outside parameter tolerance. Within a 90-percent confidence interval, the average range factor for 11,000 feet was 2,069 ±6 NAM. For 13,500 feet, the average range factor was 2,027 ±9 NAM. Treated testing was terminated when additional test points did not significantly change the range factor values or confidence intervals.

Average Range Factor Data Comparison:

Qualitative analysis revealed a small improvement in average standardized range factor for the treated aircraft at both altitudes, however, no statistical significance could be assigned to the results due to the instrumentation uncertainty. At 11,000 feet, the difference between baseline (2,055 ±6 NAM) and treated (2,069 ±6 NAM) average range factors was 14 ±12 NAM, or 0.7 ±0.6 percent. At 13,500 feet, the difference between the baseline (2,020 ±5 NAM) and treated (2,027 ±9 NAM) average range factors was 7 ±14 NAM, or 0.3 ±0.7 percent. Table 5 summarizes the average range factor comparison.

Table 5
AVERAGE RANGE FACTOR[1] COMPARISON

- T-38A S/N 68-8153
- J85-GE-5 engines
- Mach number = 0.77
- W/δ = 17,600 pounds
- cg = 18 ±0.1 percent MAC
- JP-8 fuel
- Lower Heating Value = 18,636 Btu/lb

	Pressure Altitude (ft)	
	11,000	13,500
Baseline (NAM)	2,055 ±6[2]	2,020 ±5
Treated, (NAM)	2,069 ±6	2,027 ±9
Difference (NAM)	14 ±12	7 ±14
Difference[3] (pct)	0.7 ±0.6	0.3 ±0.7

Note: NAM - nautical air miles

[1] Range factor = (true airspeed) X (aircraft gross weight)/(fuel flow)
[2] Tolerances represent 90-percent confidence intervals
[3] Difference = (treated - baseline)/baseline * 100

Error analysis determined the maximum instrumentation uncertainty in range factor to be ±0.6 percent (Appendix E). Figure 1 shows the average range factor data with instrumentation uncertainty error bars for the baseline and treated aircraft at each standard altitude. This instrumentation uncertainty was greater than the observed change in range factor, therefore, it was not possible to determine within a 90-percent confidence interval, whether or not a change in range factor attributable to Racer's Edge polish was evident. **The test should be re-accomplished with a DAS having a better instrumentation uncertainty. (R1)**[3]

Additional Observations:

Flight test data showed that the calculated instrumentation uncertainty was correct. As statistical confidence approached 99.9 percent, the confidence interval for the range factor data approached ±0.6 percent, which was equal to the calculated instrumentation uncertainty.

The HAVE SLICKER test results corresponded with hydraulic smoothness theory presented in Appendix E and previous tests (Reference 7). The test team found an average range factor dependence on altitude. This was primarily caused by two factors, both attributable to decreasing Reynolds number with altitude. First, there was an increase in skin friction drag due to a lower Reynolds number.[4] Reynolds number was defined as the ratio of airflow momentum to airflow viscosity. As Reynolds number decreased, the viscous effects became more pronounced, thus increasing skin friction drag. Second, there was a decrease in fuel efficiency due to lower engine component efficiencies. The decrease in average range factor for the T-38A aircraft was approximately 0.7 percent per thousand feet of altitude. This was comparable to previous F-16 and B-52 performance tests (Reference 7).

[3] Numerals preceded by an R within parentheses at the end of a paragraph correspond to recommendation numbers tabulated in the Conclusions and Recommendations section of this report.

[4] $R_N = \frac{Ul}{\nu}$ where:

U = true airspeed,
l = characteristic length, in this case, wing chord, and
ν = kinematic viscosity

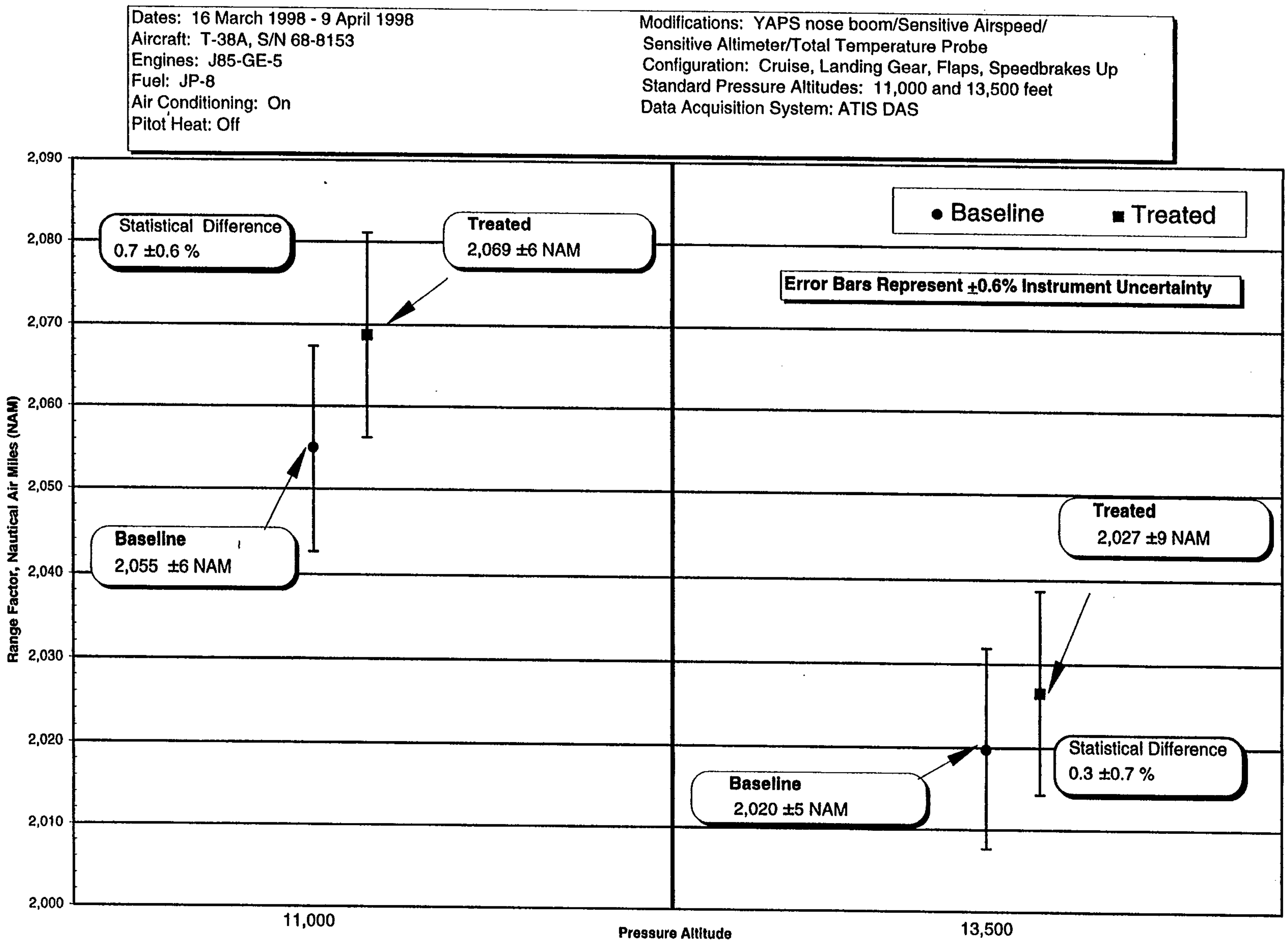

Figure 1 Standard Range Factor at Two Pressure Altitudes (11,00 feet and 13,500 feet)

CONCLUSIONS AND RECOMMENDATION

The overall test objective of characterizing the range factor (true airspeed multiplied by aircraft gross weight divided by fuel flow) change as a result of an application of the Racer's Edge polish to a T-38A aircraft was met. While flight test results revealed a trend toward improved range factors at both altitudes, the improvement was less than the ±0.6-percent instrumentation uncertainty. Test results revealed a 0.7-percent improvement in the range factor at 11,000 feet pressure altitude and a 0.3-percent improvement at 13,500 feet pressure altitude; therefore, a 90-percent confidence interval in a measurable improvement in the range factor was not attained at either altitude, and no meaningful statistical confidence could be obtained.

1. The test should be re-accomplished with a data acquisition system having a better instrumentation uncertainty. (Page 10)

REFERENCES

1. Tennent, Scott G., Captain, USAF, *A Limited Evaluation of a Surface Treatment for Aircraft Drag Reduction (HAVE SLICK)*, AFFTC-TLR-95-38, Air Force Flight Test Center, Edwards AFB, California, June 1995.

2. Flight Manual, USAF Series Aircraft, T-38A, Technical Order 1T-38A-1, San Antonio ALC/TILT, Kelly AFB, Texas, 1 July 1987, change 10, 1 June 1997.

3. Modification Flight Manual, T-38 S/N: 68-153, 412 OG/OGV, Edwards AFB, California, 7 October 1997.

4. USAF Test Pilot School Instrumentation Handbook, USAF TPS/TSF, Edwards AFB, California, June 1996.

5. USAF Test Pilot School Performance Phase Planning Guide, USAF TPS/ED, Edwards AFB, California, July 1997.

6. Basic Weight Checklist & Loading Data, USAF Series Aircraft, T-38A, Technical Order 1T-38A-5, San Antonio ALC/TILT, Kelly AFB, Texas, 30 Aug 1996.

7. Olsen, Wayne. June 1982. *Aircraft Performance Short Course Notes Volume I.* AFFTC Flight Dynamics Branch, Edwards AFB, California.

8. *USAF TPS Cruise Textbook*, USAF TPS/ED, Edwards AFB, California, 1991.

9. Hayes, B.R., *Flight Manual Performance USAF Model T-38A Trainer Airplane with Two J85-GE-5 Engines*, NOR-60-350, Northrop Corporation, NORAIR Division, 1960.

10. Walski, W.F., *Flight Test Performance Analysis of the Northrop T-38A with Two J85-GE-5 Engines Supersonic Basic Trainer*, NOR-62-34, Northrop Corporation, NORAIR Division, 1962.

11. Perry, L.J., *Standard Aircraft Characteristics Performance of the Northrop T-38A Supersonic Basic Trainer with Two J85-GE-5 Engines*, NOR-62-16, Northrop Corporation, NORAIR Division, 1962.

12. Schlichting, Hermann, Dr. 1960. *Boundary Layer Theory.* New York: McGraw-Hill Book Company, Inc.

APPENDIX A

PROGRAM INTRODUCTION DOCUMENT

PROGRAM TITLE: HAVE SLICKER

CUSTOMER DATA:

a. Requesting Agency:

American Aviation and Toolcraft
6034 Country Lane
Quartz Hill, CA 93526

b. Project Representative: Larry Sweetser (808) 823-0733

c. Statement of Capability Desired Date: ASAP

d. Other support agencies:

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T Brennan, Inc. (818) 363-5300

Support Agency:
AFFTC/XPST (POC: Kurt Buehler)

PROGRAM IDENTIFICATION INFORMATION:

Begin Date:	7 Jan 98
First Test Date:	9 Feb 98 (TPWG)
Completion date:	10 Jun 98

SYSTEM BACKGROUND INFORMATION: The Racer's Edge product is a unique, water soluble, thin-film, polish/wax. Applications of this product makes the surface smoother reducing the friction factor. Early investigative studies show possible benefit in aircraft glide ratios, lift-to-drag ratios, and decreasing parasitic drag. There is potential for improved fuel economies. The coating will not change the appearance of the plane. There is limited build-up (1 micron) when the product is applied and the product will not degrade the surface. This product is also non-toxic, biodegradable and requires no HAZMAT devices nor clothing nor protocol (Ref: AFFTC Bio-environmental Material Health and Safety Bulletin).

TEST PROGRAM AND MISSION INFORMATION/OBJECTIVES: The objective of this test is to provide initial flight test measurements of the aerodynamic drag reduction capabilities of the Racer's Edge polish (provided by American Aviation & Toolcraft).

ENVIRONMENTAL CONSIDERATIONS: See material and safety bulletin from the AFFTC bio-environmental office.

ACTIVITY PLAN: American Aviation & Toolcraft will provide the Racer's Edge product to be tested, as well as supervise application on the surface of an instrumented T-38 (Tail # 153 - verify tail # w/ Shadow - Joe Everett). The test planning, test conduct, data reduction, data analysis, and reporting will use current technology TPS test assets. It is anticipated that approximately 10 sorties will be required to accomplish the test. Initial sorties will be flown without the polish so as to provide a basis for comparison.

SYSTEM INFORMATION: T-38 (instrumented, Tail # 153). Racer's Edge Polish.

ELECTRONIC/ELECTRO-OPTICAL SYSTEM INFORMATION: N/A

INSTRUMENTATION SYSTEMS: N/A

TELEMETRY/DATA RANGE: N/A

AIR/GROUND COMMUNICATIONS: N/A

DATA PROCESSING/DISPLAY/CONTROL: AFFTC/TS in the person of Mr. Frank Brown to provide limited consultation support.

PHOTOGRAPHIC SUPPORT: At the cost and discretion of TPS.

METEOROLOGICAL: N/A

RECOVERY: N/A

OTHER TECHNICAL SUPPORT: N/A

MEDICAL: Standard medical support for TPS sorties.

PUBLIC AFFAIRS SERVICES: N/A

BASE FACILITIES/LOGISTICS: N/A

SERVICES REQUIRED:

Fire and rescue: Standard TPS support
Security and safety: N/A
Community Education and Food Service: N/A
Utilities: Standard TPS support
Air Conditioning and Environmental: N/A
Physical and/or Life Science Experiments: N/A
Propellants, Gases, and Chemicals: N/A
Fuels and Lubricants, Hydraulic Fluids. Preservatives, etc.: Standard
Requesting Agency Aircraft: N/A
Air Operations: Standard flight test support

Sea Craft: N/A
Marine Operations: N/A
Chemical Cleaning: N/A

LABORATORY: N/A

MAINTENANCE PLANNING: The only services required will be general aircraft maintenance as required. However, assuming the polish is applied at Edwards, the team of 4 personnel from T Brennan Inc. responsible for applying the polish will need access to Hangar 1600. Procedure takes roughly 5 hours.

MANPOWER AND PERSONNEL: N/A

SUPPLY SUPPORT: N/A

SUPPORT EQUIPMENT: Standard ground support equipment for a T-38.

TECHNICAL DATA: Suggested reference for details concerning potential methods for data acquisition and reduction: HAVE SLICK Test Report (POC Mr Dave Lazerson).

TRAINING REPORT: N/A

COMPUTER RESOURCES: N/A

FACILITIES: A hangar is needed for use during the application of the polish

PACKAGING, HANDLING, STORAGE, AND TRANSPORTATION: N/A

MODIFICATION: Addition of the polish will be considered a mod note (preferred) or a Class 2 Mod. TPS POC: Sharlene Lim (ext 3410).

SPECIALTY ENGINEERING: N/A

MATERIAL HEALTH AND
SAFETY BULLETIN

MANUFACTURER'S NAME
RACER'S EDGE PRODUCTS

STREET ADDRESS
19431 BUSINESS CENTER DR. # 30

CITY, STATE AND ZIP CODE
NORTHRIDGE CALIFORNIA 91324

EMERGENCY PHONE NUMBER (24 Hours):
Transportation Emergencies Call: CHEMTREC (800) 424-9300
Health Emergencies Call: Los Angeles Poison Information Center (213) 664-2121

PRODUCT: AUTO POLISH

CHEMICAL NAME:

CAS NUMBER: (Not Applicable for Blends)

DOT (Proper Shipping Name)

HAZARD RATING NFPA

EAST	FIRE	-	1
SLIGHT	TOXICITY	-	2
2-MODERATE	REACTIVITY	-	0
3-HIGH	SPECIAL	-	
4-EXTREME			

WARNING STATEMENT:

DANGER: Contains Petroleum distillated and morpholine. Harmful if swallowed. If swallowed DO NOT induce vomiting. Call a Physician immediately. Avoid eye contact and prolonged skin contact.

USE IN WELL VENTILATED AREAS.
KEEP OUT OF THE REACH OF CHILDREN
FOR INDUSTRIAL USE ONLY.

SECTION I .. INGREDIENTS

PRODUCT	CAS NUMBER	TLV	PEL	PERCENTAGES
Mineral Spirits Stoddard Solvents)	64741-41-9	100A	100 ppm	10-12%
Parafins, Cycloparafins and Aromatics		NE	NE	10-15%
Isopropanol	67-63-0	400A	400 ppm	2-5%
Turpentine	800-66-42	100A	100 ppm	2-5%
Morpholine	110-91-8	20B	20 ppm	½-1%
Water Emulsion Blend: Polysiloxane Mixture	9016-00-6	NE	NE	60-70%

Threshold Limit Value

A, Osha [x] B, ACGIH [x] C, See Section III [ ] D, Other [ ] Cal Osha

Section II .. EMERGENCY AND FIRST AID PROCEDURES

FMERGENCY: Have a physician call: LOS ANGELES POISON INFORMATION CENTER (24 Hrs.) (213) 644-2121

EYE CONTACT	Gently flush with large quantities of water for at least 15 minutes. Seek medical attention immediately.
SKIN CONTACT	Remove any contaminated clothing. Wash with soap and large quantities of water. Seek medical attention if irritated.
NHALATION	If breathing difficulties, dizziness, or light-headedness occur when working in areas with high vapor concentration, move to outside air immediately. If breathing stops, begin artificial respiration and seek immediate medical attention.
INGESTION	If this product is swallowed, seek medical attention immediately. DO NOT induce vomiting unless directed by a physician.

Section III .. PHYSIOLOGICAL EFFECTS AND HEALTH INFORMATION

YE FFECTS	This product may be an eye irritant.
KIN FFECTS	Prolonged skin contact may result in irritation and/or Dermatitis.
YSTEMIC FFECTS	Various studies have shown a possible association with exposure to this product and the following: NONE

RCINOGEN: NTP IARC MONOGRAPHS OSHA

NONE KNOWN

RACERS EDGE AVIATION POLISH

Racers Edge Polymer Aviation Polish protects all non-porus substrates from premature deterioration and destruction due to the particular environment that the substrate is subjected to Typical substrates for which such protection is desired includes metal or composite surfaces, painted, clearcoated or otherwise with any combination of paints including acrylic, enamel or lacquer . Applying the polymer polish to these materials forms a barrier between the surface of the substrate and the environment to protect the underlying material from deterioration which, if left unchecked could, result in the decline of the aesthetic appearance as well as damage.

By providing a protective film on the surface of the treated material, the polymer polish prevents materials from bonding or sticking to the surface of the treated substrate. The polymer polish which contains 2-propenoic acid-2-cynao-3, 3-diphenyl-2-ethylhexyl ester provides for surface bonding and the high gloss polymer protective layers for enhancement of the surface's color. Furthermore, it will not discolor over time. The polish contains new liquid ultraviolet light absorbers from the hydroxypheylbenzotriazole class. This inhibitor is unique of it high thermal stability and thermal permanence. The polish provides 100% aabsorptionover the UV spectrum.

Significantly, the polymer polish restores the original luster to the dull or faded surfaces by the interaction of the polish with the substrate itself. The polymer polish removes oxidation on the surface and then adheres to the substrate itself through polar bonding or entrapment due to the color absorbent particles in the polish. Critical application is therefore not required because of the bonding properties. Application of a thin layer to the surface will suffice to enable the polymers to adhere.

The backbone of the polymer polish is a unique blend of polysiloxanes which through the polycondensation chemical reaction form hydrophobic (water sheeting) and extremely hard coating films which are superior to classical wax and cleaner formulations

The rheological additives provide the polymer polish composition with the desire elasticity, viscosity and plasticity allowing for ease of use and mistake free application. It will clean oxidized surfaces, hard water spots, mineral residue, and oils from the surface and spread the polish evenly as it smooths the silicone layers. It also helps to buff the treated surface and causes color from a treated surface to blend. Because of the silicone polymers unique to the polish and the rheological additives a single coating of the polish should provide durability of the protective coating for 6 to 12 months.

APPENDIX B

T-38A, USAF S/N 68-8153, CALIBRATION RECORDS

T-38 ON AIRCRAFT INSTRUMENTATION CALIBRATION VALIDATION

TAIL NUMBER: 153 DATE: 10 MAR 98

FRONT COCKPIT TTU-205 S/N		Altitude S/N 9815			Airspeed S/N 11029				Mach S/N 14731		
Alt (ft)	Airspeed (knots)	Up	Down	Hyst	Up	Down	Hyst	Mach	Up	Down	Hyst
2300 ±50	150 ±5	2301	2305	±25	149	149.5	±2		/ /	/ /	
5000 ±50	300 ±5	5000	5010	±25	300	300.5	±2	0.49 ±.02	/ /	/ /	±.02
10000 ±75	450 ±8	10010	10025	±30	449	449.5	±3	0.80 ±.02	.805	.805	±.02
15000 ±75	600 ±8	14985	15000	±30	596	596	±3	1.14 ±.02	1.145	1.145	±.02
20000 ±75	500 ±8	19965	19975	±30	501	500	±3	1.04 ±.02	1.045	1.045	±.02
25000 ±100	400 ±5	24950	24955	±40	398	398	±3	0.93 ±.02	.935	.93	±.02
30000 ±100	350 ±5	29980	29990	±40	349.5	349	±3	0.91 ±.02	.905	.905	±.02
35000 ±150	250 ±5	35025	35030	±40	249	248.5	±3	0.74 ±.02	.735	.735	±.02
40000 ±150	200 ±5	40100	40105	±40	198.5	198	±3	0.67 ±.02	.66	.66	±.02

Low Speed Accuracy Test

Tester @ 3300ft	IND
200 Kts	198.5
190	189
180	178.5
170	168.5
160	159
150	149.5

Leak Check (PACER A/C 154 & 574)

15,000 ft ≤ 50 ft/min	N/A
300 Kts ≤ 1 Kt/min	

Use the measurement after the third one minute check.

TESTER LEAK CHECK

@15,000 ft	8 ft
@300 kts	.2 KTS

Leak Check TPS

15000 ft. ≤ 100 ft/min	56
40000 ft. ≤ 200 ft/min	184
200 kts. ≤ 3 kts/min	2.3
600 kts. ≤ 4 kts/min	.3 KT

T-38 153 10 MAR 98

Ray G. Julio P.

TPS Cockpit Instrumentation Validation\ Revision 1.1\ Jul 96

FICP
[illegible] 89163

INSTALLED F/C/P
A/c 153
3-10-98

INSTRUMENT CALIBRATION DATA REDUCTION
412 TEST WING / TSIS
EDWARDS AIR FORCE BASE, CALIFORNIA
(805) 275-4356
DSN 525-4356

Nomenclature : AIRSPEED		Work Order # :	45834
Type / Model : MOD-650		Requestor :	WINTER
Part Number : 739U-03		Calibrated By :	NAKATA
Serial Number : 11029		Press.Amb :	27.758
Manufacturer : KOLLSMAN		Temp.Amb :	22.78

DATE CAL : 16 DEC 1997
REMARKS : CAL PER TPS SPEC.
STD USED : 3(211-993-047)

*** RAW DATA LISTING ***

Input Units : KNOTS
Output Units : KNOTS

Pnt	Input	Reading Up	Correct Up	Reading Down	Correct Down	Hysteresis	Specs Fail
1	50.000	50.000	0.000	50.000	0.000	0.000	
2	60.000	59.000	1.000	59.000	1.000	0.000	
3	70.000	69.000	1.000	69.500	0.500	0.500	
4	80.000	80.000	0.000	81.000	-1.000	1.000	
5	90.000	93.000	-3.000	93.000	-3.000	0.000	
6	100.000	101.000	-1.000	99.000	1.000	2.000	
7	110.000	110.000	0.000	110.000	0.000	0.000	
8	120.000	120.000	0.000	120.000	0.000	0.000	
9	130.000	130.000	0.000	130.000	0.000	0.000	
10	140.000	140.000	0.000	140.000	0.000	0.000	
11	150.000	149.000	1.000	149.000	1.000	0.000	
12	160.000	158.000	2.000	159.000	1.000	1.000	
13	170.000	168.000	2.000	168.000	2.000	0.000	
14	180.000	178.000	2.000	178.000	2.000	0.000	
15	190.000	188.000	2.000	189.000	1.000	1.000	
16	200.000	198.000	2.000	198.000	2.000	0.000	
17	210.000	208.000	2.000	208.000	2.000	0.000	
18	220.000	218.000	2.000	218.000	2.000	0.000	
19	230.000	228.000	2.000	228.500	1.500	0.500	
20	240.000	238.000	2.000	239.000	1.000	1.000	
21	250.000	248.000	2.000	248.500	1.500	0.500	
22	260.000	258.000	2.000	259.000	1.000	1.000	
23	270.000	268.000	2.000	269.000	1.000	1.000	
24	280.000	279.000	1.000	279.000	1.000	0.000	
25	290.000	289.000	1.000	290.000	0.000	1.000	
26	300.000	300.000	0.000	300.000	0.000	0.000	
27	310.000	310.000	0.000	310.000	0.000	0.000	
28	320.000	320.000	0.000	320.000	0.000	0.000	
29	330.000	330.000	0.000	330.000	0.000	0.000	
30	340.000	339.500	0.500	340.000	0.000	0.500	
31	350.000	349.000	1.000	350.000	0.000	1.000	
32	360.000	358.000	2.000	359.000	1.000	1.000	
33	370.000	368.000	2.000	369.000	1.000	1.000	
34	380.000	378.000	2.000	378.500	1.500	0.500	
35	390.000	388.000	2.000	388.500	1.500	0.500	
36	400.000	398.000	2.000	399.000	1.000	1.000	

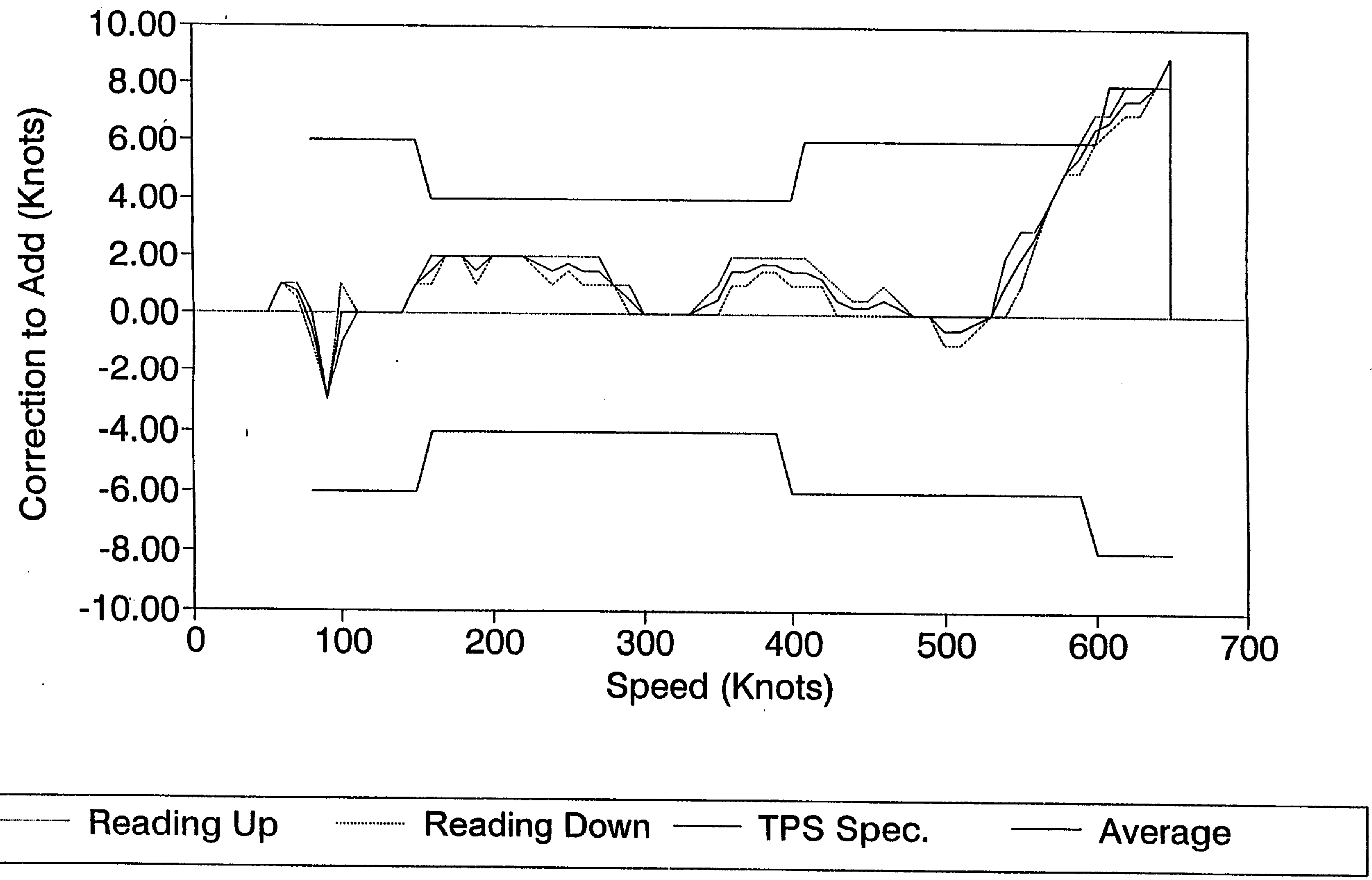
Work Order # : 45834
erial # : 11029

RAW DATA LISTING
Input Units : KNOTS
Output Units : KNOTS

Pnt	Input	Reading Up	Correct Up	Reading Down	Correct Down	Hysteresis	Specs Fail
37	410.000	408.000	2.000	409.000	1.000	1.000	
38	420.000	418.500	1.500	419.000	1.000	0.500	
39	430.000	429.000	1.000	430.000	0.000	1.000	
40	440.000	439.500	0.500	440.000	0.000	0.500	
41	450.000	449.500	0.500	450.000	0.000	0.500	
42	460.000	459.000	1.000	460.000	0.000	1.000	
43	470.000	469.500	0.500	470.000	0.000	0.500	
44	480.000	480.000	0.000	480.000	0.000	0.000	
45	490.000	490.000	0.000	490.000	0.000	0.000	
46	500.000	500.000	0.000	501.000	-1.000	1.000	
47	510.000	510.000	0.000	511.000	-1.000	1.000	
48	520.000	520.000	0.000	520.500	-0.500	0.500	
49	530.000	530.000	0.000	530.000	0.000	0.000	
50	540.000	538.000	2.000	540.000	0.000	2.000	
51	550.000	547.000	3.000	549.000	1.000	2.000	
52	560.000	557.000	3.000	557.500	2.500	0.500	
53	570.000	566.000	4.000	566.000	4.000	0.000	
54	580.000	575.000	5.000	575.000	5.000	0.000	
55	590.000	584.000	6.000	585.000	5.000	1.000	
56	600.000	593.000	7.000	594.000	6.000	1.000	TPS
57	610.000	603.000	7.000	603.500	6.500	0.500	
58	620.000	612.000	8.000	613.000	7.000	1.000	
59	630.000	622.000	8.000	623.000	7.000	1.000	
60	640.000	632.000	8.000	632.000	8.000	0.000	
61	650.000	641.000	9.000	641.000	9.000	0.000	TO/TPS

AIRSPEED CALIBRATION

12/16/1997 Work Order: 45834 Serial Number: 11029



R|C|P

T38/153

T-38 ON AIRCRAFT INSTRUMENTATION CALIBRATION VALIDATION

10 MAR 98

REAR COCKPIT		Altitude S/N 9255			Airspeed S/N 11931				Mach S/N 15105		
Alt (ft)	Airspeed (knots)	Up	Down	Hyst	Up	Down	Hyst	Mach	Up	Down	Hyst
2300 ±50	150 ±5	2290	2300	±25	150	151.5	±2	—	/ /	/ /	
5000 ±50	300 ±5	4,980	4,980	±25	298	299	±2	0.49 ±.02	/ /	/ /	±.02
10000 ±75	450 ±8	9,990	10,000	±30	448	449	±3	0.80 ±.02	.8	.8	±.02
15000 ±75	600 ±8	14,980	14,995	±30	598	598	±3	1.14 ±.02	1.15	1.15	±.02
20000 ±75	500 ±8	19,995	20,000	±30	499	497	±3	1.04 ±.02	1.05	1.05	±.02
25000 ±100	400 ±5	24,950	24,960	±40	399	398	±3	0.93 ±.02	.93	.93	±.02
30000 ±100	350 ±5	29,970	29,990	±40	349	348.5	±3	0.91 ±.02	.91	.91	±.02
35000 ±150	250 ±5	34,970	34,990	±40	249.5	249	±3	0.74 ±.02	.74	.74	±.02
40000 ±150	200 ±5	39,960	39970	±40	200	200	±3	0.67 ±.02	.66	.66	±.02

Low Speed Accuracy Test

Tester @ 3300ft	IND
150 Kts	151.5
160	160.5
170	170.5
180	180.5
190	190
200	200

TPS Cockpit Instrumentation Validation\ Revision 1.1\ Jul 96

R/QP
removed 3027

R/C/P
AK 153
03-10-98

INSTRUMENT CALIBRATION DATA REDUCTION
412 TEST WING / TSIS
EDWARDS AIR FORCE BASE, CALIFORNIA
(805) 275-4356
DSN 525-4356

Nomenclature	: AIRSPEED	Work Order #	: 45782
Type / Model	: MOD-650	Requestor	: WINTER
Part Number	: 739BU-03	Calibrated By	: NAKATA
Serial Number	: 11931	Press.Amb	: 27.758
Manufacturer	: KOLLSMAN	Temp.Amb	: 22.78

DATE CAL : 4 DEC 1997
REMARKS : CAL PER TPS SPEC.
STD USED : 3(211-993-047)

*** RAW DATA LISTING ***

Input Units : KNOTS
Output Units : KNOTS

Pnt	Input	Reading Up	Correct Up	Reading Down	Correct Down	Hysteresis	Specs Fail
1	50.000	54.000	-4.000	55.000	-5.000	1.000	
2	60.000	62.000	-2.000	64.000	-4.000	2.000	
3	70.000	72.000	-2.000	74.000	-4.000	2.000	
4	80.000	83.000	-3.000	85.000	-5.000	2.000	
5	90.000	94.000	-4.000	96.000	-6.000	2.000	TO/TPS
6	100.000	102.500	-2.500	103.000	-3.000	0.500	
7	110.000	111.000	-1.000	112.000	-2.000	1.000	
8	120.000	120.000	0.000	121.000	-1.000	1.000	
9	130.000	130.000	0.000	131.500	-1.500	1.500	
10	140.000	141.000	-1.000	142.000	-2.000	1.000	
11	150.000	150.000	0.000	152.000	-2.000	2.000	
12	160.000	160.000	0.000	162.000	-2.000	2.000	
13	170.000	171.000	-1.000	171.500	-1.500	0.500	
14	180.000	180.000	0.000	181.500	-1.500	1.500	
15	190.000	190.000	0.000	191.000	-1.000	1.000	
16	200.000	200.000	0.000	200.000	0.000	0.000	
17	210.000	210.000	0.000	210.000	0.000	0.000	
18	220.000	220.000	0.000	220.000	0.000	0.000	
19	230.000	229.000	1.000	230.000	0.000	1.000	
20	240.000	239.000	1.000	240.000	0.000	1.000	
21	250.000	249.000	1.000	249.000	1.000	0.000	
22	260.000	259.000	1.000	260.000	0.000	1.000	
23	270.000	269.500	0.500	270.000	0.000	0.500	
24	280.000	279.500	0.500	280.000	0.000	0.500	
25	290.000	289.000	1.000	289.000	1.000	0.000	
26	300.000	298.000	2.000	298.500	1.500	0.500	
27	310.000	308.000	2.000	308.000	2.000	0.000	
28	320.000	317.500	2.500	318.000	2.000	0.500	
29	330.000	327.000	3.000	328.000	2.000	1.000	
30	340.000	337.500	2.500	338.000	2.000	0.500	
31	350.000	348.000	2.000	348.500	1.500	0.500	
32	360.000	358.000	2.000	359.000	1.000	1.000	
33	370.000	369.000	1.000	370.000	0.000	1.000	
34	380.000	379.000	1.000	380.000	0.000	1.000	
35	390.000	388.500	1.500	389.000	1.000	0.500	
36	400.000	398.000	2.000	399.000	1.000	1.000	

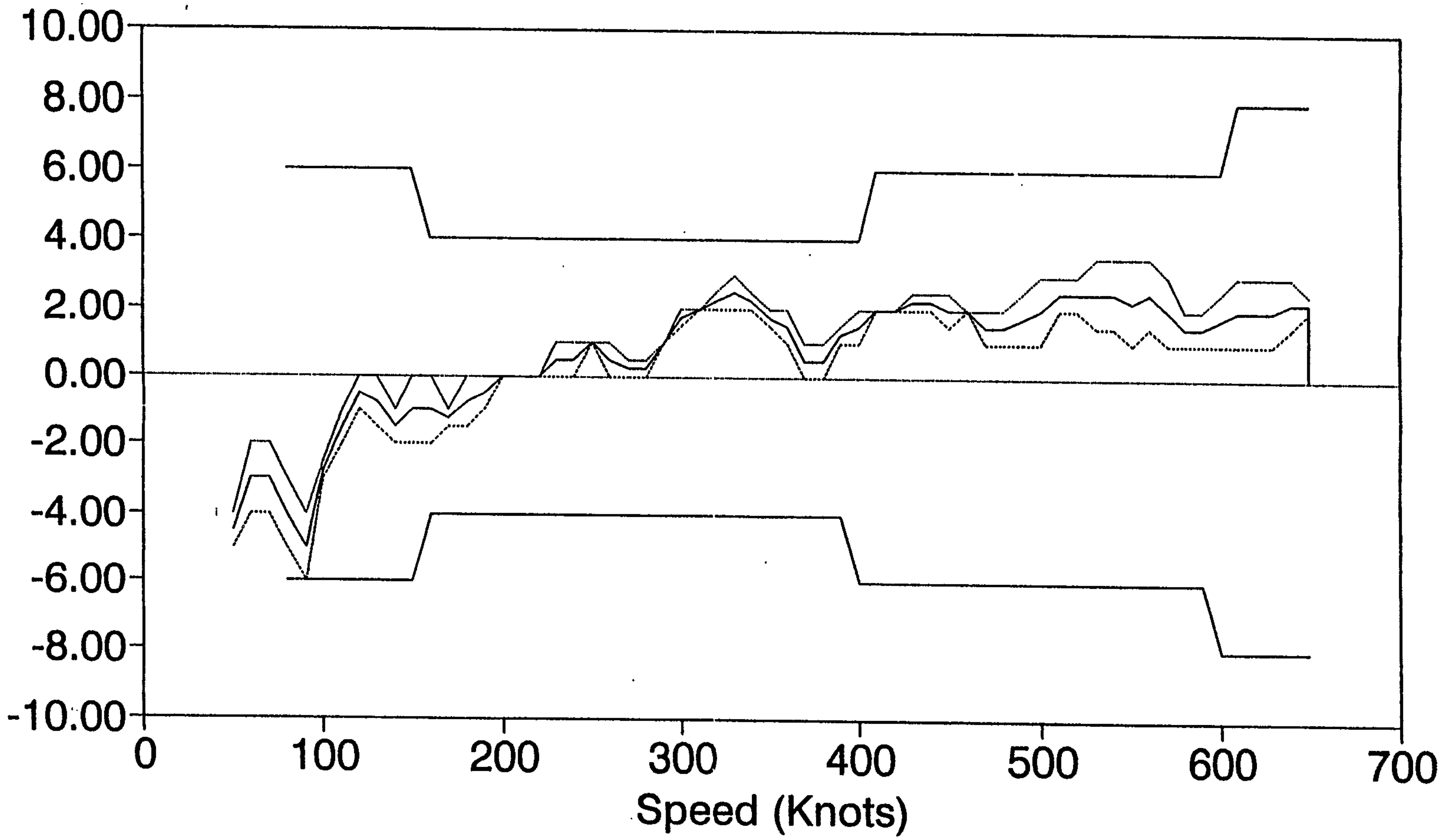
Work Order # : 45782
erial # : 11931

RAW DATA LISTING
Input Units : KNOTS
Output Units : KNOTS

Pnt	Input	Reading Up	Correct Up	Reading Down	Correct Down	Hysteresis	Specs Fail
37	410.000	408.000	2.000	408.000	2.000	0.000	
38	420.000	418.000	2.000	418.000	2.000	0.000	
39	430.000	427.500	2.500	428.000	2.000	0.500	
40	440.000	437.500	2.500	438.000	2.000	0.500	
41	450.000	447.500	2.500	448.500	1.500	1.000	
42	460.000	458.000	2.000	458.000	2.000	0.000	
43	470.000	468.000	2.000	469.000	1.000	1.000	
44	480.000	478.000	2.000	479.000	1.000	1.000	
45	490.000	487.500	2.500	489.000	1.000	1.500	
46	500.000	497.000	3.000	499.000	1.000	2.000	
47	510.000	507.000	3.000	508.000	2.000	1.000	
48	520.000	517.000	3.000	518.000	2.000	1.000	
49	530.000	526.500	3.500	528.500	1.500	2.000	
50	540.000	536.500	3.500	538.500	1.500	2.000	
51	550.000	546.500	3.500	549.000	1.000	2.500	
52	560.000	556.500	3.500	558.500	1.500	2.000	
53	570.000	567.000	3.000	569.000	1.000	2.000	
54	580.000	578.000	2.000	579.000	1.000	1.000	
55	590.000	588.000	2.000	589.000	1.000	1.000	
56	600.000	597.500	2.500	599.000	1.000	1.500	
57	610.000	607.000	3.000	609.000	1.000	2.000	
58	620.000	617.000	3.000	619.000	1.000	2.000	
59	630.000	627.000	3.000	629.000	1.000	2.000	
60	640.000	637.000	3.000	638.500	1.500	1.500	
61	650.000	647.500	2.500	648.000	2.000	0.500	

AIRSPEED CALIBRATION

12/4/1997 Work Order: 45782 Serial Number: 11931



Reading Up ...... Reading Down —— TPS Spec. —— Average

A/C TYPE:T-38A TAIL#: 8153 DATE:12MAR98

INSTRUMENT INSTALLATION DATES A/S:10MAR98 ALT:10MAR98
INSTRUMENT SERIAL NUMBERS A/S:11029 ALT:9815

INSTRUMENTATION CORRECTIONS

VI	DVIC	HI	DHIC	TI	DTIC
50	0.0	0	0.0	-50	0.0
60	1.0	1000	7.5	-46	0.0
70	0.75	1500	2.5	-42	0.0
80	-0.5	1600	7.5	-38	0.0
90	-3.0	1800	5.0	-34	0.0
100	0.0	2000	-10.0	-30	0.0
110	0.0	2200	-10.0	-26	0.0
120	0.0	2400	0.0	-22	0.0
130	0.0	2600	0.0	-18	0.0
140	0.0	2800	-2.5	-14	0.0
150	1.0	3000	-17.5	-10	0.0
160	1.5	4000	-15.0	-6	0.0
170	2.0	5000	-15.0	-2	0.0
180	2.0	6000	-25.0	2	0.0
190	1.5	7000	-22.5	6	0.0
200	2.0	8000	-30.0	10	0.0
210	2.0	9000	-35.0	14	0.0
220	2.0	10000	-35.0	18	0.0
230	1.75	11000	-30.0	22	0.0
240	1.5	12000	-30.0	26	0.0
250	1.75	13000	-22.5	30	0.0
260	1.5	14000	-10.0	34	0.0
270	1.5	15000	-5.0	38	0.0
280	1.0	16000	5.0	42	0.0
290	0.5	17000	15.0	46	0.0
300	0.0	18000	15.0	50	0.0
310	0.0	19000	15.0	54	0.0
320	0.0	20000	15.0	58	0.0
330	0.0	22000	15.0	62	0.0
340	0.25	24000	22.5	66	0.0
350	0.5	26000	30.0	70	0.0
360	1.5	28000	12.5	74	0.0
370	1.5	30000	-15.0	78	0.0
380	1.75	32000	-15.0	82	0.0
390	1.75	34000	-77.5	86	0.0
400	1.5	36000	-100.0	90	0.0
410	1.5	38000	-145.0	94	0.0
420	1.25	40000	-172.5	98	0.0
430	0.5	42000	-165.0		
440	0.25	44000	-145.0		
450	0.25	46000	-110.0		
460	0.5	48000	-65.0		
470	0.25	50000	-12.5		
480	0.0				
490	0.0				
500	-0.5				
510	-0.5				
520	-0.25				
530	0.0				
540	1.0				
550	2.0				
560	2.75				
570	4.0				

580	5.0
590	5.5
600	6.5
610	6.75
620	7.5
630	7.5
640	8.0
650	9.0

T-38 PARAMETERS (DAS Measurands)

PARAMETER NAME	SOURCE	RANGE	RESOLUTION	APPROXIMATE MEASUREMENT UNCERTAINTY	SAMPLES PER SEC
Right Engine Fuel Flow	Transducer	0.5 to 10 GPM	0.05 GPM	0.05 GPM	4
Right After Burner Fuel Flow	Transducer	15 to 25 GPM	0.05 GPM	0.05 GPM	4
Right Fuel Used	Transducer - 20 Bit	0 to 2000 gal	0.01 gal	0.1 GPM *	4
Left Engine Fuel Flow	Transducer	0.5 to 10 GPM	0.05 GPM	0.05 GPM	4
Left After Burner Fuel Flow	Transducer	15 to 25 GPM	0.05 GPM	0.05 GPM	4
Left Fuel Used	Transducer - 20 Bit	0 to 2000 gal	0.01 gal	0.1 GPM *	4
Event Counter	Transducer	0 to 99 Count	Discrete	N/A	4
Event Marker	Transducer	0 or 1	Discrete	N/A	32
Longitudinal Stick Force	Transducer	±70 lbs	0.17 lbs	0.85 lbs	32
Lateral Stick Force	Transducer	±35 lbs	0.08 lbs	0.5 lbs	32
Left Rudder Pedal Force	Transducer	0 to -150 lbs	0.15 lbs	0.75 lbs	32
Right Rudder Pedal Force	Transducer	0 to 150 lbs	0.15 lbs	0.75 lbs	32
θ - Pitch Angle	Transducer	±70°	0.06°	0.3°	32
ϕ - Roll Angle	Transducer	±175°	0.35°	1.75°	32
p - Roll Rate	Transducer	±360°/sec	0.7°/sec	3.5°/sec	32
q - Pitch Rate	Transducer	±20°/sec	0.05°/sec	0.25°/sec	32
r - Yaw Rate	Transducer	±20°/sec	0.05°/sec	0.25°/sec	32
Total Pressure	Transducer	0.4 to 38 PSIA	0.0002 PSIA	0.0038 PSIA	32

~~6-17~~

T-38 PARAMETERS, (DAS Measurands)

PARAMETER NAME	SOURCE	RANGE	RESOLUTION	APPROXIMATE MEASUREMENT UNCERTAINTY	SAMPLES PER SEC
Static Pressure	Transducer	0.4 to 38 PSIA	0.0002 PSIA	0.0038 PSIA	32
Right Engine RPM	Transducer	25 to 102 %RPM	0.15%	0.75%	32
Left Engine RPM	Transducer	25 to 102 %RPM	0.15%	0.75%	32
α - Angle of Attack	Transducer	-10 to 30°	0.04°	0.2°	32
β- Angle of Sideslip	Transducer	±20°	0.04°	0.2°	32
Right Engine Fuel Temp	Transducer	-50° to 150° C	0.3° C	1.5° C	32
Left Engine Fuel Temp	Transducer	-50° to 150° C	0.3° C	1.5° C	32
Total Air Temperature	Transducer	-55° to 85° C	0.11° C	0.5° C	32
Normal Acceleration	Transducer	-3 to 6 g	0.01 g	0.05 g	32
Lateral Acceleration	Transducer	±1 g	0.002 g	0.01 g	32
Longitudinal Acceleration	Transducer	±1 g	0.002 g	0.01 g	
Longitudinal Stick Position	Transducer	-4 to 7 in	0.02 in	0.1 in	32
Lateral Stick Position	Transducer	±8 in	0.02 in	0.1 in	32
Rudder Pedal Position	Transducer	±3 in	0.01 in	0.05 in	32
Stabilator Position	Transducer	-6° to 16°	0.03°	0.15°	32
Right Aileron Position	Transducer	-25° to 35°	0.08°	0.4°	32
Left Aileron Position	Transducer	-35° to 25°	0.08°	0.4°	32
Rudder Position	Transducer	±30°	0.07°	0.35°	32
IRIG Time					32
Hot Mike					

*** Fuel Used is derived from Fuel Flow. It's error is therefore a function of time.**

APPENDIX C

RANGE FACTOR CALCULATIONS

RANGE FACTOR CALCULATIONS

GENERAL

The T-38A data acquisition system (DAS) was used to collect data, and the Test Pilot School (TPS) Aydin was used to extract the data from the DAS ¼-inch tape. These data were loaded into a spreadsheet for data reduction. The theory and equations were all from the USAF TPS Cruise textbook (Reference 8).

The start and end values for pressure altitude (Hc), M, and ambient air temperature (T_a) were obtained from the DAS. The average value for each test point was derived. From the DAS pressure altitude, the test team derived the ambient air pressure ratio (δ_t) with the following equation:

$$\delta_t = (1 - 6.87558(10)^{-6} Hc)^{5.2559} \qquad \text{(C1)}$$

The standard altitude ambient air pressure ratio (δ_s) was likewise found by:

$$\delta_s = (1 - 6.87558(10)^{-6} Hcs)^{5.2559} \qquad \text{(C2)}$$

using the standard day pressure altitude (Hc_s). The ambient air temperature ratio (θ_t) was found from:

$$\theta_t = \frac{T_a}{288.15} \qquad \text{(C3)}$$

where Ta was in degrees Kelvin.

The standard altitude ambient air temperature ratio (θ_s) was likewise found by:

$$\theta_s = \frac{T_{as}}{288.15} \qquad \text{(C4)}$$

using the standard day ambient air temperature (T_{as}) for that pressure altitude. The standardized average normalized fuel flow ($\dot{W}_{fs}$) was then defined as:

$$\dot{W}_{fs} = \dot{W}_{ft} \frac{\delta_s \sqrt{\theta_s}}{\delta_t \sqrt{\theta_t}} \qquad \text{(C5)}$$

where:

$\dot{W}_{fs}$ = standardized, normalized average fuel flow, and

$\dot{W}_{ft}$ = test day average fuel flow, corrected for fuel heating value. This was simply the measured fuel flow times the fuel heating value ratio as determined by Phillips Lab.

From this, the test team computed the standardized specific range (SR_s):

$$SR_s = \frac{Ma_o \sqrt{\theta_s}}{\dot{W}_{fs}} \qquad \text{(C6)}$$

where:

SR_s = standardized specific range, NAM per pound,

M = Mach number, and

A_o = speed of sound, sea level standard day, 661.48 NAM per hour.

Then from this, the test team computed range factor (RF) with:

$$RF = SR_s W_s \qquad \text{(C7)}$$

where:

W_s = standard weight.

CORRECTIONS TO DATA

The HAVE SLICK (Reference 1) program showed specific range improvements of about 1 percent. The anticipated change in range factor due to the polish was expected to be equally small (1 to 2 percent). In order to maximize the likelihood of detecting this small change, standardization to a reference set of conditions (pressure altitude and

M) and corrections for specific excess thrust and specific range due to M variations were applied to the data. Data from each 3-minute data point were reduced to a single average data point.

To minimize trim drag effects, all points were flown as closely as possible to the same cg (18 percent mean aerodynamic chord [MAC]). However, due to lack of available data on trim drag effects due to cg, no corrections were made for actual cg variation from 18 percent MAC.

Standardization To A Reference Set Of Conditions:

For the statistical sample, the test team estimated that at least 25 baseline and 25 treated data points had to be compared at the same flight conditions. The aircraft gross weight to ambient air pressure ratio (W/δ) flight test technique permitted the data to be gathered at a constant W/δ. As fuel was consumed (decreasing aircraft gross weight) a constant W/δ was maintained by flying at progressively at higher altitudes (decreasing ambient air pressure).

Correction For Specific Excess Thrust:

The cruise test point began with a trim shot, where the desired test parameters were set, including throttle position. A small error in throttle setting would cause an excess thrust (P_s) which would cause a change in altitude, velocity, or a combination of both. Once the test point started, small climbs or descents were corrected with minor pitch adjustments. However, any velocity changes were dealt with postflight by correcting for the specific excess thrust. By definition,

$$P_S = \dot{H} + \frac{V_t}{g}\dot{V}_t \qquad \text{(C8)}$$

where:

$\dot{H}$ = time rate of change of tapeline altitude, feet per second,

V_t = true airspeed, feet per second,

$\dot{V}_t$ = time rate of change of true airspeed, feet per second squared, and longitudinal flight path acceleration (n_x), measured in g's was:

$$n_x = \frac{P_S}{V_t} \qquad \text{(C9)}$$

Since excess thrust (F_{ex}) was defined by:

$$F_{ex} = n_x GW \qquad \text{(C10)}$$

where:

GW = aircraft average gross weight for the test point.

Excess fuel flow (FF_{ex}) was defined in terms of F_{ex} and thrust specific fuel consumption (TSFC) by:

$$FF_{ex} = F_{ex}(TSFC) \qquad \text{(C11)}$$

For the T-38A, let TSFC = 1.00 lb/hr/lb.

Making the appropriate substitutions, the test team arrived at:

$$FF_{ex} = \frac{(P_S)(GW)(TSFC)}{V_t} \qquad \text{(C12)}$$

This value was subtracted from the measured fuel flow value to account for excess power to give corrected fuel flow (FF_{corr}):

$$FF_{corr} = FF_{test} - FF_{ex} \qquad \text{(C13)}$$

Corrections For Specific Range:

Mach number error, in the context of this report, was the change in drag due to flying a test point at an off-target M. Any variation in M would change the actual drag experienced by the aircraft. This slight change in M would cause a change in the drag and, therefore, fuel flow. To correct for M variations, the test team used specific range data presented in Northrop Corporation NORAIR Reports NOR 60-350 and NOR 62-34 (References 9 and 10). These reports contained flight test and wind tunnel derived aerodynamic coefficients and aircraft performance estimates. Specific range versus M plots gave a local slope at the test M. This local slope was used to find a specific range correction due to the error in M. The measured specific range was corrected and then used to find a corrected range factor.

APPENDIX D

TEST POINTS AND FLIGHT ENVELOPE

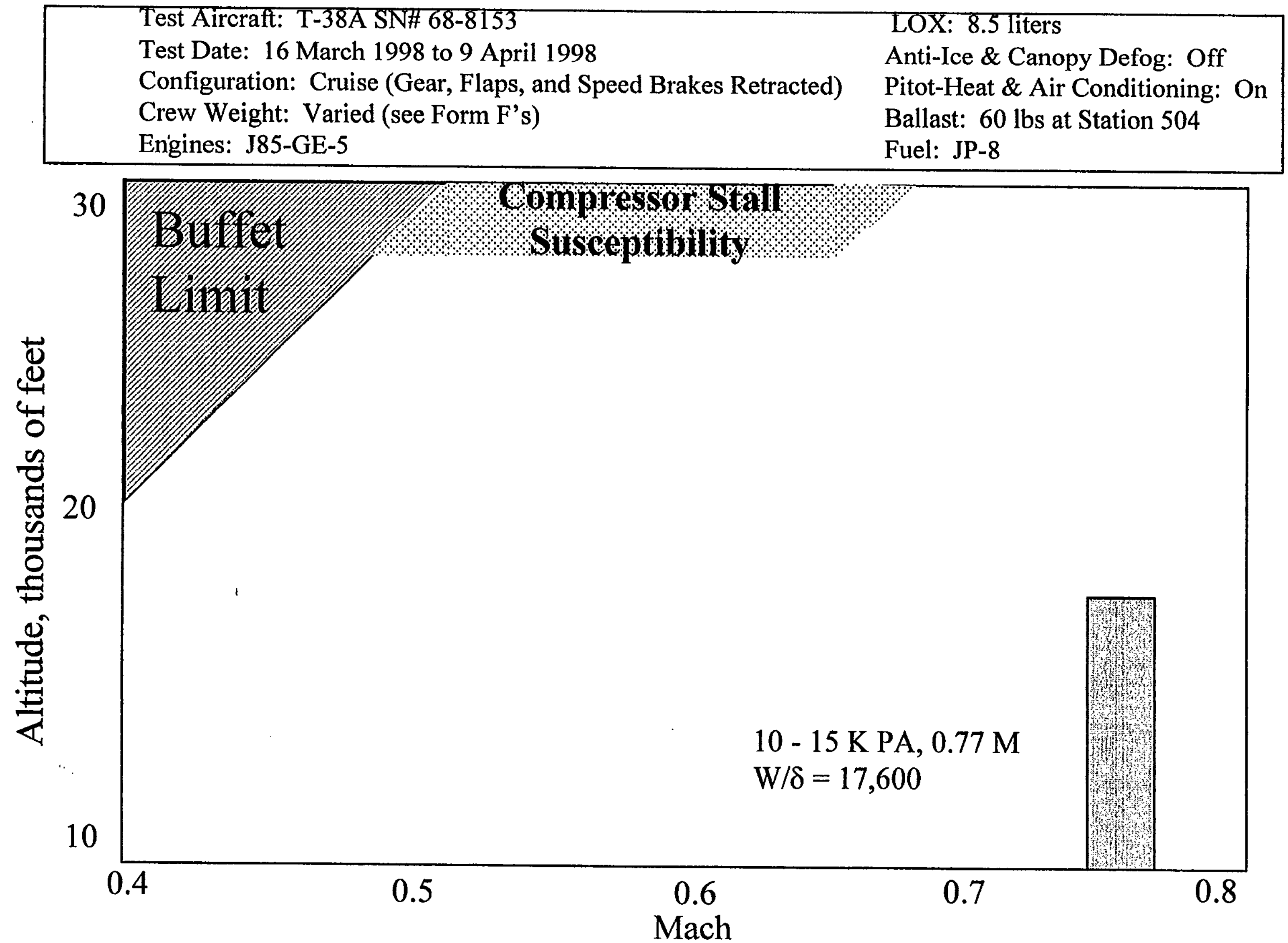


Figure D1 HAVE SLICKER Test Point and Flight Envelope

APPENDIX E

ERROR ANALYSIS AND TEST POINT SELECTION

ERROR ANALYSIS AND TEST POINT SELECTION

GENERAL

The basis for a change in range factor due to an application of Racer's Edge polish was a change in drag. The test team did not have the capability to accurately measure the drag of the test aircraft. However, range factor was determined from measurable quantities available from the currently available instrumentation system. This appendix gives a general discussion of aircraft drag and its relationship to range factor, as well as quantifying drag changes, the drag savings mechanism, selecting the test conditions, and corrections to the data.

AIRCRAFT DRAG AND RANGE FACTOR

Aircraft Drag:

Total aircraft drag (C_D) could be expressed as:

$$C_D=C_{D_i}+C_{D_f}+C_{D_p}+C_{D_b}$$

where:

C_{D_i} = coefficient of drag due to lift,
C_{D_f} = coefficient of skin friction drag,
C_{D_p} = coefficient of pressure drag, and
C_{D_b} = coefficient of base drag.

Northrop conducted both wind tunnel and flight tests to determine the drag values throughout the flight envelope (Reference 11). Table E1 summarizes the composition of the different drag components at one of the flight conditions (W/δ=17,600 pounds, gross weight = 12,500 to 9,500 pounds, Mach number = 0.77, pressure altitude = 9,500 to 17,500 feet).

Range Factor:

Range factor was calculated from the following:

$$RF=SR*GW \qquad \text{(E1)}$$

where:

RF = range factor, nautical air miles (NAM),
SR = specific range (NAM/lb), and
GW = aircraft gross weight, (lb),

$$SR=V_t/FF_{ec} \qquad \text{(E2)}$$

where:

V_t = true airspeed, (kt),
FF_{ec} = energy corrected fuel flow, (lbs/hr), and

$$FF_{ec}=FF_m*FHVR \qquad \text{(E3)}$$

where:

FF_m = measured volumetric fuel flow, (lb/hr),
FHVR = fuel heat value ratio, (dimensionless),

$$FF_m=FF_v*r_f(F_t) \qquad \text{(E4)}$$

where:

FF_v = measured volumetric fuel flow, (gal/hr), and
r_f (F_t)= fuel density as a function of fuel temperature, F_t, (lb/gal).

The energy corrected fuel flow (FF_{ec}) was the measured volumetric fuel flow (FF_v) times the fuel density (a function of fuel temperature) all corrected for variations in fuel heat value ratio (FHVR).

AIRCRAFT DRAG AND RANGE FACTOR RELATIONSHIP

For each 3-minute stable point, the following assumptions were made:

1. Thrust and drag were constant,

2. Thrust equals drag after correcting for energy height through excess thrust corrections,

3. Volumetric fuel flow was constant,

4. Fuel temperature was constant,

5. Aircraft gross weight was constant, and

6. Thrust was a function of mass fuel flow.

Table E1
DRAG COEFFICIENTS

- Lift Coefficient = 0.14
- Mach Number = 0.72
- Pressure Altitude = 13,500 ft

Component of Drag	Coefficient	Percent of Total
Drag due to Lift, C_{D_i}	0.00200	12.2
Skin Friction Drag, C_{D_f}	0.01200	73.4
Pressure Drag, C_{D_p}	0.00250	15.3
Base Drag, C_{D_b}	-0.00015	-0.9
Total	0.01635	100.0

From these assumptions and the previous equations, it was assumed that: range factor (RF) was inversely proportional to drag.

The test team concluded that changes in range factor were inversely proportional to changes in drag. Therefore, changes in range factor were used as the basis for determining changes in drag after application of the polish.

QUANTIFYING DRAG CHANGES

Accurate, absolute values of range factor, or drag, were not critical; only relative changes were required to show an improvement. Relative changes in range factor were used to quantify any changes in drag.

REDUCING SKIN FRICTION DRAG

Test point selection was made on the assumption that an improvement in range factor would most likely come in the form of reduced skin friction drag. Two methods existed for reducing skin friction drag. The first was to delay the transition from laminar to turbulent boundary layer. However, since the T-38A aircraft was already sufficiently smooth and the flight Reynolds number was high, the transition point was most likely fixed by the pressure gradient and free-stream turbulence. The other method was to reduce the turbulent boundary layer skin friction drag by reducing the surface roughness.

Protuberances Outside the Laminar Sublayer:

At the test flight conditions, the boundary layer over the entire T-38A was almost completely turbulent (Reference 12). According to Schlichting (Reference 12) "in the case of the turbulent boundary layer roughness has no effect, and the wall is hydraulically smooth if all protuberances are contained within the laminar sublayer." However, Schlichting (Reference 12) also stated "in most practical applications connected with the flat plate (e.g. ships, lifting surfaces of an aircraft, turbine blades) the wall cannot be considered hydraulically smooth." Therefore, the T-38A aircraft was not hydraulically smooth and had some roughness associated with protuberances outside the laminar sublayer of the turbulent boundary layer.

Admissible Roughness:

According to Schlichting (Reference 12):

> the amount of roughness which is considered "admissible" in engineering applications was that maximum height of individual roughness elements which caused no increase in drag compared to a smooth wall.

In layman's terms, the admissible roughness was the maximum height of surface elements that did not increase drag over that of a perfectly smooth wall. For practical purposes, if a surface had a roughness larger than the admissible roughness (protuberances outside the laminar sublayer), the drag increased. If the roughness was less than the admissible roughness (hydraulically smooth), then the drag was no more than that of a perfectly smooth surface.

Figure E1 shows a hydraulically smooth surface. Notice how the surface protuberances lay within the laminar sublayer, thus, was not contributing to the drag of the hydraulically smooth surface. Figure E2 shows a surface with increased drag due to surface roughness. Notice that the surface protuberances extended beyond the region of the laminar sublayer, thereby increasing skin friction drag.

In order to characterize a reduction in skin friction drag due to a polish application, the aircraft was operated in a region where the admissible roughness was smaller than the roughness of the surface. This caused the baseline test flights to be conducted with the protuberances penetrating through the laminar sublayer, into the turbulent boundary layer. For drag reduction to occur, the polish must reduce the height of the protuberances (by filling in the gaps between the protuberances, effectively reducing their height above the skin surface) such that they did not penetrate beyond the laminar sublayer. The roughness, or average root mean squared height of the surface protuberances, of the T-38A aircraft was measured as 0.26 mils (0.00026 inches). The admissible roughness was independent of the length of the surface; it was determined solely by the velocity and kinematic viscosity (i.e., it was a function of the Reynolds number):

$$K_{adm} \leq 100 * \nu / U_{\infty} \qquad \text{(E5)}$$

where:

ν = viscosity, and
U_{∞} = the free stream velocity.

In order to obtain an admissible roughness less than 0.00026 inches, the aircraft was operated in a region of high freestream velocity and low viscosity (the lower right corner of the flight envelope). A plot of admissible roughness is shown in Figure E3.

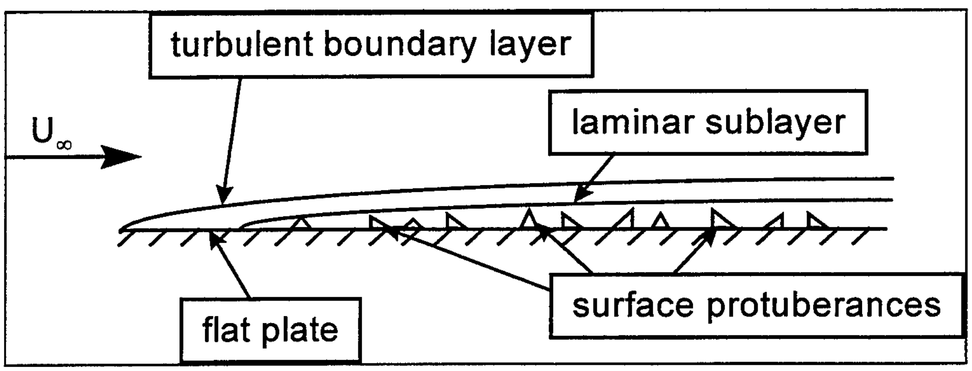


Figure E1 Hydraulically Smooth Surface

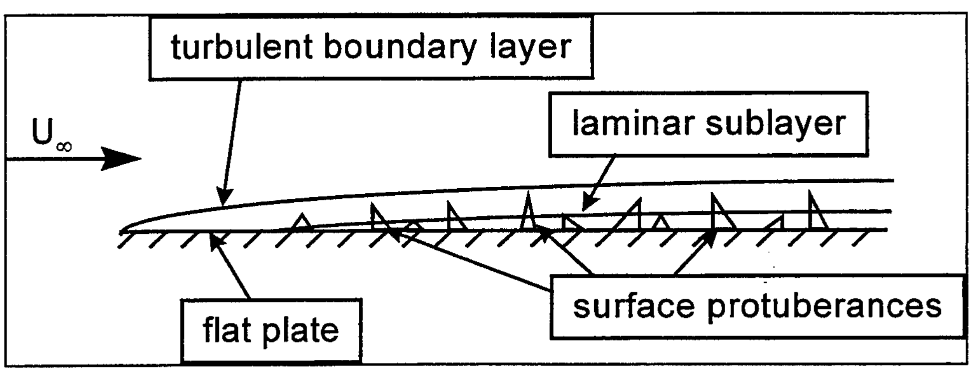


Figure E2 Nonhydraulically Smooth Surface

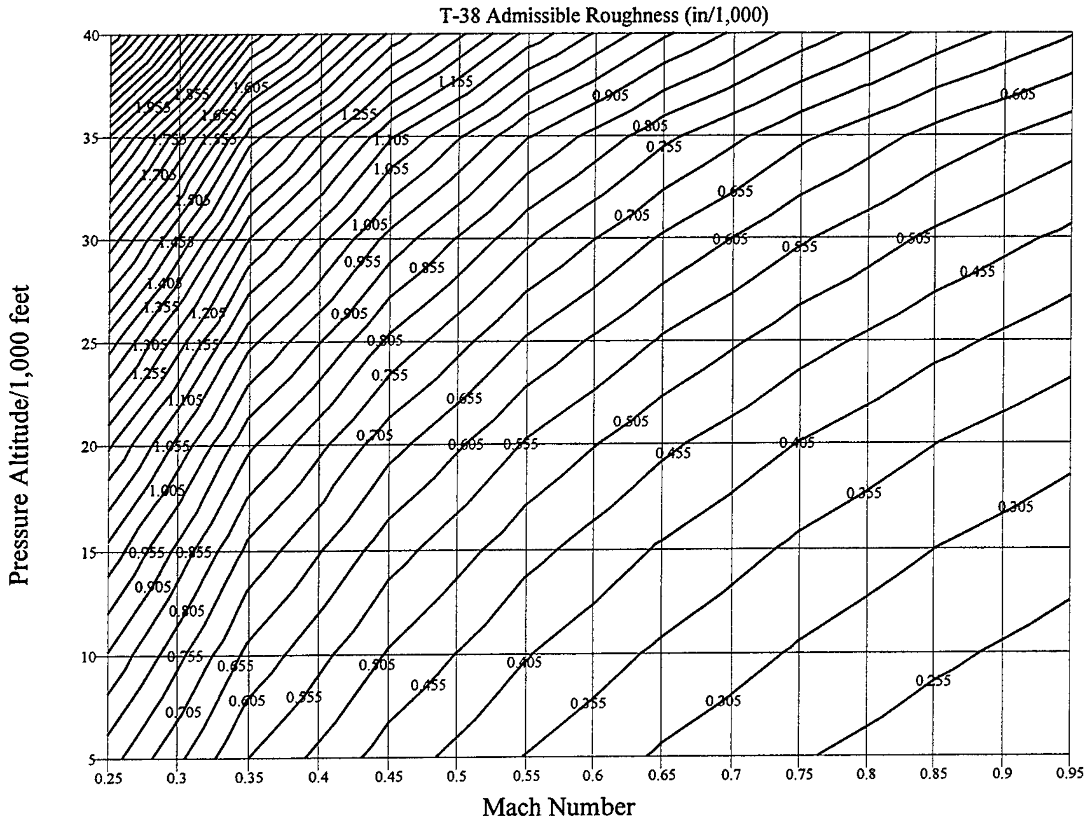


Figure E3 Plot of Admissible Roughness

SELECTING THE TEST CONDITIONS

Selecting the flight test W/δ and Mach number involved both the admissible roughness and expected variations in drag due to airspeed variations. First, the test point had to be where the admissible roughness was less than the baseline roughness of the T-38A aircraft. Therefore, all of the test W/δ data points had to have an admissible roughness of 0.00026 inches or less. When this was true, the protuberances were larger than the laminar sublayer of the boundary layer. A W/δ of 17,600 pounds at 0.77 Mach number satisfied these conditions.

Sensitivity Analysis:

A sensitivity analysis was performed to estimate the uncertainties in range factor between the baseline flights and the treated flights. The analysis was performed using the following equation:

$$Error_{RMS} = \Delta s = \sqrt{\left(\frac{\partial s}{\partial u_1}\Delta u_1\right)^2 + \left(\frac{\partial s}{\partial u_2}\Delta u_2\right)^2 + \ldots} \qquad \text{(E6)}$$

This equation defined the uncertainty in s as Δs, where s was a function of u_1, u_2, etc. Repeated applications of Equation 6 were made on Equations 7 through 28 using the conservative representative values listed in Tables E2 through E15. (If a range of values for a parameter was available, the most conservative values were used to ensure the uncertainty was worst case.)

Mach Number:

$$QPsl = (Pt-P)/14.696 \quad (E7)$$

$$QPs = QPsl/(P/14.696) \quad (E8)$$

$$M=[[(QPs+1)^{\wedge}(0.287514)-1]/0.2]^{\wedge}0.5 \quad (E9)$$

where:

P_t = total pressure from the data acquisition system (DAS), psi,
P = pressure from DAS (psi), and
M = Mach number.

Table E2
MACH NUMBER DERIVATION AND UNCERTAINTY VALUES

Parameter	Expected Value	Source of Data	Uncertainty	Source of Uncertainty
P_t	12.4500 psi	Data acquistion system (DAS)	0.0038 psi	DAS
P	8.8110 psi	DAS	0.0038 psi	DAS
QPsl	0.2476	Equation 7	0.0004	Equation 6
QPs	0.4130	Equation 8	0.0006	Equation 6
M	0.7229	Equation 9	0.0005	Equation 6

Notes:
1. P_t - total pressure from the DAS, (psi)
2. P - pressure from DAS, psi
3. QPsl - dynamic pressure at sea level
4. QPs - dynamic pressure (psi)
5. M - Mach number

Fuel Density:

$$\rho = \rho s + m*[T_f - T_{fs}] \quad (E10)$$

where:

ρ = fuel density, (lb/gal),
ρs = fuel sample density, (lb/gal),
m = slope of the fuel density versus fuel temperature curve, (lb/gal)/deg F,
T_f = fuel temperature, (deg F), and
T_{fs} = fuel sample temperature, (deg F).

Table E3
FUEL DENSITY DERIVATION AND UNCERTAINTY VALUES

Parameter	Expected Value	Source of Data	Uncertainty	Source of Uncertainty
ρ	6.905 lb/gal	Equation 10	0.0226 lb/gal	Equation 6
ρs	6.839 lb/gal	Phillips Lab	0.01 lb/gal[1]	Phillips Lab
m	-0.0033 (lb/gal)/deg F	Phillips Lab	0.001(lb/gal)/deg F	Equation 6
T_f	40.1 deg F	Data acquisition system (DAS)	2.7 deg F	DAS
T_{fs}	60.1 deg F	Phillips Lab	0.90 deg F[1]	Phillips Lab

Notes:
1. ρ - fuel density
2. ρs - fuel sample density
3. m - slope of the fuel density versus fuel temperature curve
4. T_f - fuel temperature
5. T_{fs} - fuel sample temperature

[1] Estimated values

Fuel Mass:

$$FFm = FF_v * \rho \quad \text{(E11)}$$

where:

FFm = mass fuel flow, (lb/min),
FF_v = volumetric fuel flow, (gal/min), and
ρ = fuel density, (lb/gal).

Table E4
MASS FUEL FLOW DERIVATION AND UNCERTAINTY VALUES

Parameter	Expected Value	Source of Data	Uncertainty	Source of Uncertainty
FFm	33.8352 lb/min	Equation 11	0.1679 lb/min	Equation 6
FF_v	4.9001 gal/min	Data acquisition system (DAS)	0.0183 gal/min	DAS

Notes: 1. FFm - mass fuel flow
2. FF_v - volumetric fuel flow
3. DAS - data acquisition system

Altitude and Hdot:

$$H = 1-(P/14.696)^\wedge 0.190255/(6.87558*10^\wedge(-6)) \quad \text{(E12)}$$

$$Hdot = (H_f-Hi)/time \quad \text{(E13)}$$

where:

H = pressure altitude, (ft),
Hdot = time rate of change in altitude, (ft/sec),
H_f = final altitude, (ft),
Hi = initial altitude, (ft), and
time = time of cruise point, (sec).

Table E5
ALTITUDE AND HDOT DERIVATION AND UNCERTAINTY VALUES

Parameter	Expected Value	Source of Data	Uncertainty	Source of Uncertainty
H	13,489 ft	Equation 12	10.830 ft	Equation 6
Hdot	1.111 ft/sec	Data Band Tolerance	0.085 ft/sec	Equation 6
H_f	13,589 ft	Data Band Tolerance	10.830 ft	Equation 6
Hi	13,389 ft	Data Band Tolerance	10.830 ft	Equation 6
time	180 sec	t(Hf) - t(Hi)	none	N/A

Notes: 1. H - pressure altitude
2. Hdot - time rate of change in altitude
3. H_f - final altitude
4. Hi - initial altitude
5. time - time of cruise point
6. N/A - not applicable

True Airspeed and Vdot:

$$T = Tr/0.995*(1+0.2*M^2)^(-1) \tag{E14}$$

$$V_t = M*38.967*T^0.5 \tag{E15}$$

$$V_t dot = (V_{tf}-V_{ti})/time \tag{E16}$$

where:

T_r = recovery temperature, (deg K),
T = ambient air temperature, (deg K),
V_t = true airspeed, (ft/sec),
$V_t$dot = time rate of change in true airspeed, ([ft/sec]/sec),
V_{tf} = final true airspeed, (ft/sec), and
V_{ti} = initial true airspeed, (ft/sec).

Table E6
TRUE AIRSPEED AND VDOT DERIVATION AND UNCERTAINTY VALUES

Parameter	Expected Value	Source of Data	Uncertainty	Source of Uncertainty
T_r	287.31 deg K	Data acquisition system (DAS)	0.5 deg K	DAS
T	261.4 deg K	Equation 14	0.411 deg K	Equation 6
V_t	769.7 ft/sec	Equation 15	0.802 ft/sec	Equation 6
$V_t$dot	0.022 ft/sec	Data Band Tolerance	0.0063 ft/sec	Equation 6
V_{tf}	771.7 ft/sec	Data Band Tolerance	0.802 ft/sec	Equation 6
V_{ti}	767.7 ft/sec	Data Band Tolerance	0.802 ft/sec	Equation 6

Notes: 1. T_r - recovery temperature
2. T - ambient air temperature
3. V_t - true airspeed
4. $V_t$dot - time rate of change in true airspeed
5. V_{tf} - final true airspeed
6. V_{ti} - initial true airspeed
7. K - Kelvin

Specific Excess Power:

$$Ps = Hdot + V_t/g*V_t dot \tag{E17}$$

where:

Ps = specific excess power, (ft/sec), and
g = acceleration due to gravity, (ft/sec^2).

Table E7
EXCESS POWER DERIVATION AND UNCERTAINTY VALUES

Parameter	Expected Value	Source of Data	Uncertainty	Source of Uncertainty
Ps	1.6372 ft/sec	Equation 17	0.17304 ft/sec	Equation 6
g	32.18 ft/sec^2	constant	none	constant

Notes: 1. Ps - specific excess power
2. g - acceleration due to gravity

Longitudinal Acceleration:

$$nx = Ps/Vt \quad \text{(E18)}$$

where:

nx = longitudinal acceleration in the flight path axes, (dimensionless).

Table E8
LONGITUDINAL ACCELERATION DERIVATION AND UNCERTAINTY VALUES

Parameter	Expected Value	Source of Data	Uncertainty	Source of Uncertainty
nx	0.002127	Equation 18	0.00022	Equation 6

Note: nx - longitudinal acceleration in the flight path axes

Gross Weight:

$$GW = Wempty + Volf*\rho - Wfuelused \quad \text{(E19)}$$

where:

GW = total gross weight at the test point, (lbs),
Wempty = aircraft empty gross weight plus crew weight, (lbs),
Volf = volumetric fuel prior to engine start, (gal), and
Wfuelused = weight of fuel used since engine start, (lbs).

Table E9
GROSS WEIGHT DERIVATION AND UNCERTAINTY VALUES

Parameter	Expected Value	Source of Data	Uncertainty	Source of Uncertainty
GW	10,774.4 lb	Equation 19	43.68 lb	Equation 6
Wempty	8,981 lb	scales	0[1]	constant
Volf	480.3 gal	fuel truck	2.0 gal	estimated
Wfuelused	1523 lb	estimate[2]	40 lb	estimated

Notes: 1. GW - total gross weight at the test point
2. Wempty - aircraft empty gross weight plus crew weight
3. Volf - volumetric fuel prior to engine start
4. Wfuelused - weight of fuel used since engine start

[1] Weight and balance stand was assumed to have zero error
[2] Estimate based on 45 minutes into flight at the average fuel flow,(FFm) (See Table E4).

Excess Thrust:

$$F_{ex} = nx*GW \quad \text{(E20)}$$

where:

F_{ex} = excess thrust, (lbs).

Table E10
EXCESS THRUST DERIVATION AND UNCERTAINTY VALUES

Parameter	Expected Value	Source of Data	Uncertainty	Source of Uncertainty
F_{ex}	22.92 lb	Equation 20	2.424 lb	Equation 6

Note: F_{ex} - excess thrust

Excess Mass Fuel Flow:

$$FF_{ex} = F_{ex}*TSFC/3600 \tag{E21}$$

where:

FF_{ex} = excess mass fuel flow, (lb/sec), and
TSFC = thrust specific fuel consumption, (1/hr).

Table E11
EXCESS MASS FUEL FLOW DERIVATION AND UNCERTAINTY VALUES

Parameter	Expected Value	Source of Data	Uncertainty	Source of Uncertainty
FF_{ex}	0.006367 lb/sec	Equation 21	0.00068 lb/sec	Equation 6
TSFC	1.0 1/hr	constant	0.01 1/hr	estimated

Notes: 1. FF_{ex} - excess mass fuel flow
2. TSFC - thrust specific fuel consumption

Correction to Mass Fuel Flow:

$$FF_{mcor} = FFm - FFex \tag{E22}$$

where:

FF_{mcor} = corrected mass fuel flow, (lb/sec).

Table E12
CORRECTION TO MASS FUEL FLOW DERIVATION AND UNCERTAINTY VALUES

Parameter	Expected Value	Source of Data	Uncertainty	Source of Uncertainty
FF_{mcor}	0.55755 lb/sec	Equation 22	0.00288 lb/sec	Equation 6

Note: FF_{mcor} - corrected mass fuel flow

Standard Day Correction to Corrected Mass Fuel Flow:

$$\delta_t = (1-6.87558*10^\wedge(-6)*H)^\wedge(5.2559) \tag{E23}$$

$$\theta_t = T/288.15 \tag{E24}$$

$$\delta_s = (1-6.87558*10^\wedge(-6)*Hs)^\wedge(5.2559) \tag{E25}$$

$$\theta_s = T_s/288.15 \tag{E26}$$

$$FF_{mcor,s} = FF_m * (\delta_s/\delta_t) * (\theta_s/\theta_t)^\wedge 0.5 \tag{E27}$$

where:

δ_t = test ambient air pressure ratio,
θ_t = test ambient air temperature ratio,
Hs = standardization altitude, (ft),
δ_s = standard altitude ambient air pressure ratio,
T_s = standardization ambient air temperature, (deg K),
θs = standard altitude ambient air temperature ratio, and
$FF_{mcor,s}$ = standardized corrected mass fuel flow, (lb/sec).

Table E13
STANDARD DAY CORRECTION TO CORRECTED MASS FUEL FLOW DERIVATION AND UNCERTAINTY VALUES

Parameter	Expected Value	Source of Data	Uncertainty	Source of Uncertainty
δ_t	0.59956	Equation 23	0.000050	Equation 6
θ_t	0.906717	Equation 24	0.001430	Equation 6
δ_s	0.59929	Equation 25	0	N/A
Hs	13,500 ft	N/A	0	Constant
θs	0.90728	Equation 26	0	N/A
T_s	261.433 deg K	atmos table	0	N/A
$FF_{mcor,s}$	0.55734 lb/sec	Equation 27	0.00088 lb/sec	Equation 6

Notes: 1. δ_t - test ambient air pressure ratio
2. θ_t - test ambient air temperature ratio
3. δ_s - standard altitude ambient air pressure ratio
4. Hs - standardization altitude
5. θ_s - standard altitude ambient air temperature ratio
6. T_s - standardization ambient air temperature
7. $FF_{mcor,s}$ - standardized, corrected mass fuel flow
8. N/A - not applicable

Fuel Heating Value Ratio:

$$FHVR = LHVt/LHVs \tag{E28}$$

where:

FHVR = fuel heat value ratio,
LHVt = lower heating value of the fuel sample, (Btu/lb), and
LHVs = standard lower heating value, (Btu/lb).

Table E14
FUEL LOWER HEATING VALUE RATIO DERIVATION AND UNCERTAINTY VALUES

Parameter	Expected Value	Source of Data	Uncertainty	Source of Uncertainty
FHVR	1.0054	Equation 28	0.00385	Equation 6
LHVt	18,500 Btu/lb	Phillips Lab	50 Btu/lb	Phillips Lab estimate
LHVs	18,400 Btu/lb	Phillips Lab	50 Btu/lb	Phillips Lab estimate

Notes: 1. FHVR - fuel heat value ratio
2. LHVt - lower heating value of the fuel sample
3. LHVs - standard lower heating value
4. Btu – British thermal unit

Range Factor:

$$RF = Vt * GW/FF_{mcor,s}/FHVR \tag{E29}$$

where:

RF = range factor, NAM.

Table E15
RANGE FACTOR DERIVATION AND UNCERTAINTY VALUES

Parameter	Expected Value	Source of Data	Uncertainty	Source of Uncertainty
RF	2, 436 NAM	Equation 29	14 NAM	Equation 6

Notes: 1. RF - range factor
2. NAM - nautical air mile

Tracking the significant digits throughout all of the calculations yields four significant digits in the range factor. Keeping two digits in the uncertainty yields an instrumentation uncertainty of 0.6 percent (14/2436*100).

APPENDIX F

WEIGHT AND BALANCE

HAVE SLICKER WEIGHT AND BALANCE TEST DATA

Table F1
FUELING DATA

Desired Fuel Weight (Dispensed/Total)	Fuel Sample Density/Temperature	Fuel Truck Counter (Indicated/Actual)	Truck Dispensed Fuel Weight (Calculated)	Truck Error (Calculated Fuel Weight vs Scale Measured Change)
Zero	--	--	--	--
900 lb/900	6.76/14 deg C	133.1/133.4	902	-16
1,300 lb/2,200	6.75/15 deg C	192.6/192.3	1,298	+18
3,500 lb/3,500	6.75/17 deg C	192.6/192.1	1,297	-1
3,982 lb/Full	6.74/19 deg C	71.5/82.8	558	--
at 25 psi		Total actual 600.6 gal	Total 4,055	--

Note: '---' – not applicable

Table F2
WEIGHT AND BALANCE DATA

Desired Total Fuel Weight (lbs)	Pitch Attitude	Aircraft Fuel Quantity Indications	Aircraft Total Fuel (Indicated)	Total Indicated Fuel vs Scale Weight Error	Aircraft Weight (Scale Reading)	Fuel Weight Change	Center of Gravity Inches/Pct MAC
Zero	Level	0/0	0	---	8,477	---	---
900	Level	390/390	780	-138	9,395	918	354.97/25.60
	5 deg	390/400	change +10	---	9,396	---	356.06/26.80
2,200	Level	990/1,160	2,150	-48	10,675	1,280	353.90/24.47
	5 deg	985/1,160	change -5	---	10,683	total 2,198	354.72/25.36
3,500	Level	1,490/2,000	3,490	-6	11,973	1,298	353.62/24.17
	5 deg	1,490/1,970	change -30	---	11,984	total 3,496	354.48/25.10
Full (3,982)	Level	1,970/2,010	4,080	+25	12,531 calc	558 (Truck)	---
						Total 4,054	

Notes: 1. '---' – not applicable
2. MAC – mean aerodynamic chord

W/Delta 17600

Fuel Gauge Errors: 0% -5% Pilot 258 lb / Engineer 224 lb

Assumptions: Pitch = 0.00 LOX 8.5 Mach = 0.77

Total Fuel	18+/- 0.1% cg Fuel Indications Imbal +/- 30	L	R	L Boost Pumps	Crossfeed	R Boost Pumps	H_i for 200# Band	H_i for 100# band	
3500	-392	1554	1849	on	NO	on	9378	9292	
3400	-382	1509	1796	on	NO	on	9588	9483	
3300	-390	1455	1753	on	NO	on	9800	9694	
3200	-400	1400	1710	on	NO	on	10013	9906	
3100	-410	1345	1667	on	NO	on	10228	10120	
3000	-412	1294	1621	on	NO	on	10444	10336	
2900	-405	1248	1570	OFF	YES	on	10662	10553	
2800	-390	1205	1515	OFF	YES	on	10881	10771	
2700	-365	1168	1456	OFF	YES	on	11102	10991	
2600	-335	1133	1394	OFF	YES	on	11325	11213	
2500	-302	1099	1331	OFF	YES	on	11549	11437	
2400	-270	1065	1268	OFF	YES	on	11774	11661	
2300	-235	1033	1204	OFF	YES	on	12002	11888	
2200	-200	1000	1140	OFF	YES	on	12231	12116	
2100	-170	965	1078	OFF	YES	on	12462	12346	
2000	-138	931	1016	OFF	YES	on	12694	12578	
1900	-110	895	955	OFF	YES	on	12929	12811	
1800	-82	859	894	OFF	YES	on	13165	13047	
1700	-53	824	833	OFF	YES	on	13403	13284	
1600	-28	786	773	OFF	YES	on	13643	13523	
1500	0	750	713	OFF	YES	on	13885	13764	
1400	25	713	653	OFF	YES	on	14128	14006	
1300	45	673	596	OFF	YES	on	14374	14251	
1200	60	630	542	on	NO	on	14622	14498	
1100	72	586	488	on	NO	on	14872	14747	
1000	75	538	439	on	NO	on	15124	14998	
900	65	483	397	on	NO (RTB)	on			

TO FLY VFR HEMISPHERIC ALTITUDE:

H_i for	E	W	E	W	E	W
W/Delta 17,600	9500 **9465**	10500 **10468**	11500 **11470**	12500 **12474**	13500 **13484**	14500 **14493**
Mean GW	12340	11871	11415	10975	10548	10134
High Fuel	3483	3008	2549	2103	1672	1254
Low Fuel	3236	2771	2320	1884	1461	1052
Imbal +/- 30	-401	-401	-287	-138	-22	64
	L R	L R	L R	L R	L R	L R
Boost Pump	on on	on on	OFF on	OFF on	OFF on	on on
Crossfeed	NO	NO	YES	YES	YES	NO
Start	1544 1842	1296 1626	1111 1366	966 1081	812 817	653 572
Imbal +/- 30	-395	-416	-326	-172	-48	51
End	1415 1730	1192 1500	1037 1220	890 945	733 692	565 463
Imbal +/- 30	-406	-387	-247	-105	4	77

Table F3 : Sample Crew-Specific Flight Card

Test Aircraft: T-38A SN# 68-8153	Anti-Ice & Canopy Defog: Off
Test Date: 16 March 1998 to 9 April 1998	Pitot-Heat & Air-Conditioning: On
Configuration: Cruise (Gear, Flaps, and Speed Brakes Retracted)	Crew Weight: 258Lb Pilot /224 Lb Engineer
Engines: J85-GE-5	Ballast: 60 lbs at Station 504
Fuel: JP-8	LOX: 8.5 liters

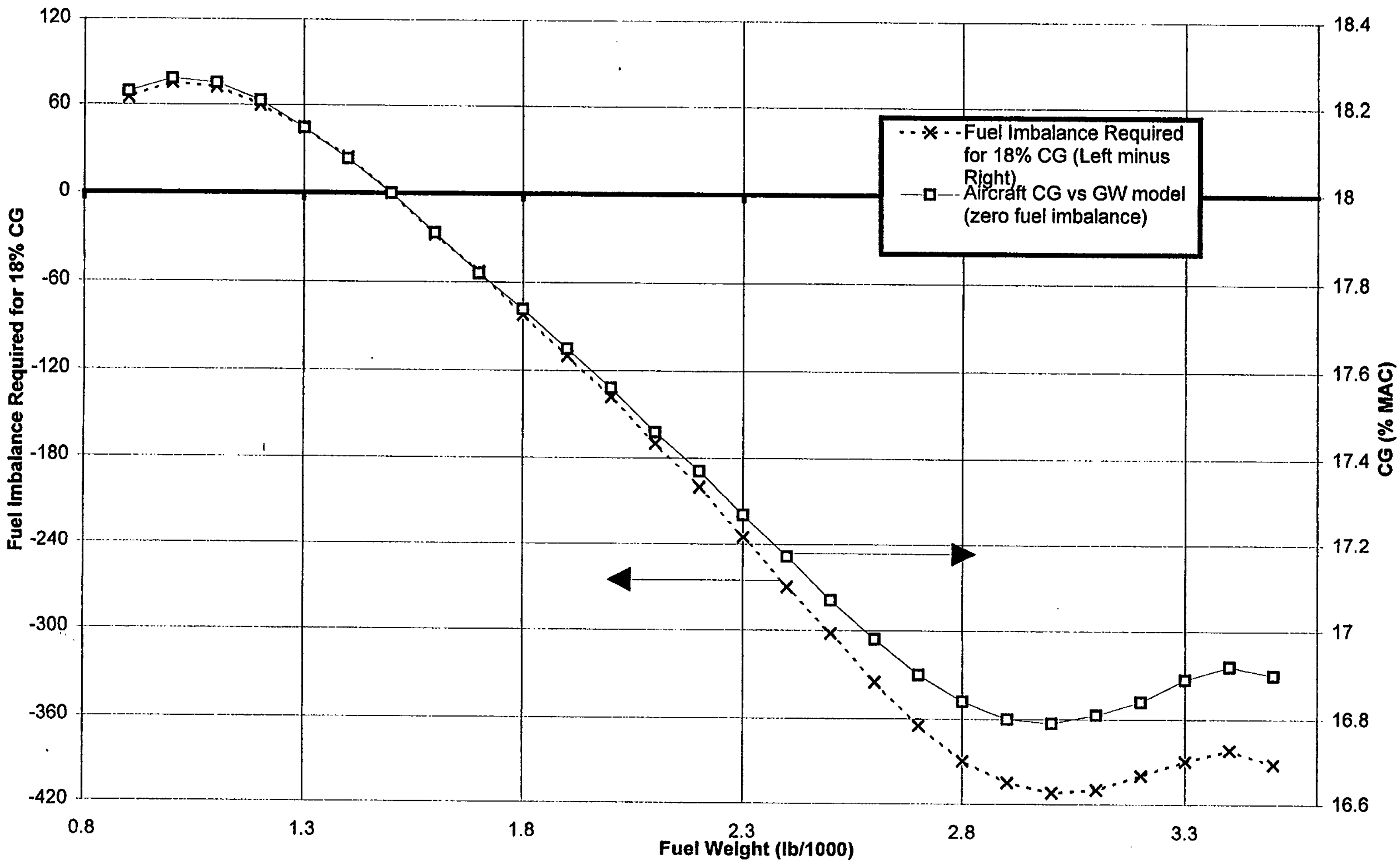


Figure F1 Sample Crew-Specific Fuel Burn Curves

FORM F - WEIGHT & BAL CLEARANCE (TACTICAL) | FORMAT: NO

DATE	MODEL/DESIGN	FROM	HOME BASE	GEAR MOM CHG
920725	T38A		EAFB,CA.	-1.6
MISSION	SERIAL NO	TO	PILOT	CONSTANT
CANNED	68-8153		FITZ	1000

REMARKS
MIN LANDING FUEL 408# REMAINING

AFT C.G. CONDITION

ITEM	WEIGHT	MOMENT
BASIC A/C From Chart C	8464	2997.4
Oil in Airplane	N/A	N/A
PILOT	258	48.8
CO-PILOT	224	56.0
LOX(10 LITERS)	26	3.3
BALLAST (1) @STA.504.0"	60	30.2
OPERATING WEIGHT	9032	3135.7

Corrections

CMPT ITEM	WEIGHT	MOMENT
SEAT PACK	-23	-5.6

Additional Load	WEIGHT	MOMENT
FUEL, INT.(583 GALS)	3964	1387.5

Less Expendables	WEIGHT	MOMENT
FUEL	3556	1240.3
[WC]	200	50.0

	WEIGHT	MOMENT
TAKEOFF (Uncorrected)	12996	4523.2
CORRECTIONS	-23	-5.6
TAKEOFF (Corrected)	12973	4517.6
LESS EXPENDABLES	3556	1240.3
LANDING (Fuel 408 lbs)	9417	3277.3

(CG in %MAC)	LIMITATIONS		2 PILOTS W/ CORR	2 PILOTS NO CORR	1 PILOT W/ CORR	1 PILOT NO CORR
TAKEOFF (GROS-WT)		13500	12973	12996	12749	12772
LANDING (GROS-WT)		13500	9417	9440	9193	9216
TAKEOFF (GEAR-DN)	14.0	25.0	18.4	18.2	20.2	20.0
LANDING (GEAR-DN)	14.0	25.0	18.1	17.9	20.7	20.4
TAKEOFF (GEAR-UP)	N/A	N/A	18.2	18.0	20.1	19.9
LANDING (GEAR-UP)	N/A	N/A	17.9	17.7	20.5	20.2
WORST CASE CONFIG	14.0	25.0	MOST FWD: 17.0		MOST AFT: 21.1	

COMPUTED BY: HPatel

WT. & BAL. AUTHORITY: [signature] 23/03/8
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PILOT

FORM F - WEIGHT & BAL CLEARANCE (TACTICAL) | FORMAT: N1

DATE	MODEL/DESIGN	FROM	HOME BASE	GEAR MOM CHG
970725	T38A		EAFB,CA.	-1.6
MISSION	SERIAL NO	TO	PILOT	CONSTANT
CANNED	68-8153		NEO	1000

REMARKS
MIN LANDING FUEL 408# REMAINING

AIRCRAFT CONDITION

ITEM	WEIGHT	MOMENT
BASIC A/C From Chart C	8464	2997.4
Oil in Airplane	N/A	N/A
PILOT	208	39.3
CO-PILOT	228	57.0
LOX(10 LITERS)	26	3.3
BALLAST (1) @STA.504.0"	60	30.2
OPERATING WEIGHT	8986	3127.2

Corrections

CMPT ITEM	WEIGHT	MOMENT
SEAT PACK	-23	-5.6

Additional Load	WEIGHT	MOMENT
FUEL, INT.(583 GALS)	3964	1387.5

Less Expendables	WEIGHT	MOMENT
FUEL	3556	1240.3
[WC]	200	50.0

	WEIGHT	MOMENT
TAKEOFF (Uncorrected)	12950	4514.7
CORRECTIONS	-23	-5.6
TAKEOFF (Corrected)	12927	4509.1
LESS EXPENDABLES	3556	1240.3
LANDING (Fuel 408 lbs)	9371	3268.8

(CG in %MAC)	LIMITATIONS		2 PILOTS W/ CORR	2 PILOTS NO CORR	1 PILOT W/ CORR	1 PILOT NO CORR
TAKEOFF (GROS-WT)		13500	12927	12950	12699	12722
LANDING (GROS-WT)		13500	9371	9594	9143	9166
TAKEOFF (GEAR-DN)	14.0	25.0	19.0	18.8	20.9	20.7
LANDING (GEAR-DN)	14.0	25.0	19.0	18.7	21.7	21.4
TAKEOFF (GEAR-UP)	N/A	N/A	18.9	18.7	20.8	20.6
LANDING (GEAR-UP)	N/A	N/A	18.8	18.5	21.5	21.2
WORST CASE CONFIG	14.0*	25.0	MOST FWD: 17.9		MOST AFT: 22.0	

COMPUTED BY: HP [illegible] WT. & BAL. AUTHORITY: [illegible signature] 23/03/8

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FORM F - WEIGHT & BAL CLEARANCE (TACTICAL) | FORMAT: N2

DATE	MODEL/DESIGN	FROM	HOME BASE	GEAR MOM CHG
970725	T38A		EAFB,CA.	-1.6
MISSION	SERIAL NO	TO	PILOT	CONSTANT
PLANNED	68-8153		LINDELL	1000

REMARKS
MIN LANDING FUEL 408# REMAINING

AFT C.G. CONDITION

ITEM	WEIGHT	MOMENT
BASIC A/C From Chart C	8464	2997.4
Oil in Airplane	N/A	N/A
PILOT	255	48.2
CO-PILOT	201	50.3
LOX(10 LITERS)	26	3.3
BALLAST (1) @STA.504.0"	60	30.2
OPERATING WEIGHT	9006	3129.4

Corrections CMPT ITEM	WEIGHT	MOMENT
SEAT PACK	-23	-5.6

Additional Load	WEIGHT	MOMENT
FUEL, INT.(583 GALS)	3964	1387.5

Less Expendables	WEIGHT	MOMENT
FUEL	3556	1240.3
[WE]	200	50.0

	WEIGHT	MOMENT
TAKEOFF (Uncorrected)	12970	4516.9
CORRECTIONS	-23	-5.6
TAKEOFF (Corrected)	12947	4511.3
LESS EXPENDABLES	3556	1240.3
LANDING (Fuel 408 lbs)	9391	3271.0

(CG in %MAC)	LIMITATIONS		2 PILOTS W/ CORR	2 PILOTS NO CORR	1 PILOT W/ CORR	1 PILOT NO CORR
TAKEOFF (GROS-WT)		13500	12947	12970	12746	12769
LANDING (GROS-WT)		13500	9391	9414	9190	9213
TAKEOFF (GEAR-DN)	14.0	25.0	18.6	18.4	20.3	20.1
LANDING (GEAR-DN)	14.0	25.0	18.4	18.2	20.8	20.5
TAKEOFF (GEAR-UP)	N/A	N/A	18.5	18.3	20.1	19.9
LANDING (GEAR-UP)	N/A	N/A	18.3	18.0	20.6	20.3
WORST CASE CONFIG	14.0	25.0	MOST FWD: 17.4		MOST AFT: 21.4	

COMPUTED BY: ARobs WT. & BAL. AUTHORITY: Kittau 23/03/8

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APPENDIX G

FUEL LOG, FUEL DENSITY, AND FUEL LOWER HEATING VALUES BY SORTIE

FUEL LOG

- T-38A S/N 69-8153
- J85-GE-5 engines
- JP-8 fuel
- Refueling pressure 25 psi

Date Serviced	Fuel Indicators (prefueling) Left (lb)	Fuel Indicators (prefueling) Right (lb)	Fuel Indicators (postfueling) Left (lb)	Fuel Indicators (postfueling) Right (lb)	Truck/Cart ID#	Quantity Serviced (gal)	Fuel Truck Fuel Temp (deg F)	Fuel Sample Temp (deg F)
19-Mar	280	210	1960	1900	90L494	522.3	66	67
23-Mar	380	360	1940	1900	91L66	470.3	76	72
24-Mar	420	200	1940	1900	91L68	469.9	70	74
24-Mar	320	270	1940	1890	90L493	506.4	74.5	76.5
27-Mar	295	270	1940	1900	90L695	502.7	57	62
27-Mar	340	270	1940	1900	90L494	502.8	61	63
30-Mar	410	270	1970	1910	90L695	480.9	56	54.5
30-Mar	300	280	1975	1900	90L695	494	65	64
2-Apr	1590	1680	1980	1910	91L69	95.6	50.5	52
2-Apr	370	280	1970	1905	91L69	485	56	57.5
3-Apr	445	405	1990	1905	91L67	433	55	62
3-Apr	245	200	1970	1900	91L69	517	64	65
6-Apr	370	300	1970	1905	90L494	484.6	55	59
7-Apr	370	270	1980	1910	90L698	500.5	55	57
8-Apr	250	270	1960	1905	90L494	510	56	60
9-Apr	650	605	1970	1910	90L494	413.8	53	60.5

Table G1: HAVE SLICKER Fuel Log

PHILLLIPS LAB RESULTS

- T-38A S/N 69-8153
- J85-GE-5 engines
- JP-8 fuel
- Refueling pressure 25 psi

Date Serviced	Low Refueling Temp (deg C)	Density (lb/gal)	Highest Temp DAS (deg C)	Density (lb/gal)	Lowest Temp DAS (deg C)	Density (lb/gal)	LHV (Btu/lb)	LHV ratio[1]
19-Mar[2]	19		38		35			
23-Mar	22	6.729	38	6.634	35	6.651	18625	0.9994
24-Mar	21	6.735	41	6.617	36	6.644	18649	1.0007
24-Mar	24	6.711	41	6.610	37	6.632	18649	1.0007
27-Mar	14	6.767	36	6.633	29	6.675	18644	1.0004
27-Mar	16	6.755	35	6.639	30	6.670	18702	1.0035
30-Mar	12	6.780	29	6.673	23	6.711	18625	0.9994
30-Mar	18	6.746	33	6.653	28	6.685	18661	1.0013
2-Apr[3]	10							
2-Apr	13	6.779	31	6.669	27	6.694	18648	1.0006
3-Apr	13	6.781	32	6.665	26	6.700	18647	1.0006
3-Apr	18	6.750	33	6.657	28	6.688	18666	1.0016
6-Apr	13	6.779	32	6.665	27	6.695	18623	0.9993
7-Apr	13	6.784	30	6.678	24	6.716	18593	0.9977
8-Apr	13	6.785	33	6.660	24	6.717	18596	0.9978
9-Apr	12	6.792	33	6.661	30	6.680	18582	0.9971

[1] Lower Heating Value Ratio=LHV/18,636 (LHV average=18,636 Btu/lb)
[2] Due to DAS failure during the sortie sample not used nor delivered to Phillips Laboratory
[3] Refueled following aborted takeoff attempt; data not recorded

Table G2: Fuel Density and Lower Heating Values

LIST OF ABBREVIATIONS, ACRONYMS, AND SYMBOLS

Abbreviation	Definition	Units
AFB	Air Force Base	---
AFFTC	Air Force Flight Test Center	---
AOA	angle of attack	units
a_o	speed of sound, sea level standard day	ft/sec
Btu	British thermal unit	---
C_D	total aircraft drag	dimensionless
C_{Db}	coefficient of base drag	dimensionless
C_{Df}	coefficient of skin friction drag	dimensionless
C_{Di}	coefficient of drag due to lift	dimensionless
C_{Dp}	coefficient of pressure drag	dimensionless
cg	center of gravity	pct MAC
DAS	data acquisition system	---
EGT	exhaust gas temperature	---
Eq	equation	---
F_{ex}	excess thrust	lb
FF_{corr}	corrected fuel flow	lb/hr
FFm_{corr}	corrected mass fuel flow	lb/hr
$FFm_{corr,s}$	standardized corrected mass fuel flow	lb/hr
FF_{ec}	energy corrected fuel flow	lb/hr
FF_{ex}	excess fuel flow	lb/hr
FF_m	mass fuel flow	lb/hr
FF_v	measured volumetric fuel flow	lb/hr
FHVR	fuel heat value ratio	dimensionless

LIST OF ABBREVIATIONS, ACRONYMS, AND SYMBOLS (Continued)

Abbreviation	Definition	Units
FTT	flight test technique	---
ft	feet	---
GW	gross weight	lb
g	acceleration due to gravity	32.2 fps^2
gal	gallon(s)	---
H	pressure altitude	ft
HAZMAT	hazardous material	---
Hc	pressure altitude	ft
Hc_s	standardized day pressure altitude	ft
Hdot, $\dot{H}$	time rate of change in altitude	feet/sec
Hi	initial altitude	ft
Hs	standardization altitude	ft
H_f	final altitude	ft
hr	hour	---
KIAS	knots indicated airspeed	---
LHVt	lower heating value	Btu/lb
LHVs	standard lower heating value	Btu/lb
lb	pound(s)	---
M	Mach number	dimensionless
MAC	mean aerodynamic chord	---
min	minute(s)	---
NAM	nautical air mile	---
n_x	longitudinal flight path acceleration	g
OAT	outside air temperature	---

LIST OF ABBREVIATIONS, ACRONYMS, AND SYMBOLS (Continued)

Abbreviation	Definition	Units
P	ambient pressure	psia
PA	pressure altitude	---
Ps	specific excess power	ft/sec
P_t	total pressure	psia
psi	pounds per square inch	---
ρ	fuel density	lb/gal
ρ_s	fuel sample density	lb/gal
RF	range factor	NAM
RPM	revolutions per minute	---
r_f	fuel density, as a function of fuel temp	lb/gal
S/N	serial number	---
SR	specific range	NAM
SR_s	standardized specific range	NAM
sec	second(s)	---
TM	trademark	---
TPS	Test Pilot School	---
TR	technical report	---
TSFC	thrust specific fuel consumption	lb/hr
T_a	ambient air temperature	deg K
T_{as}	standardized ambient air temperature	deg K
T_f	fuel temperature	deg C
T_{fs}	fuel sample temperature	deg F
T_r	recovery temperature	deg K
T_s	standard ambient air temperature	deg K

LIST OF ABBREVIATIONS, ACRONYMS, AND SYMBOLS (Concluded)

Abbreviation	Definition	Units
USAF	United States Air Force	---
U_∞	free stream velocity	ft/sec
V_t	true airspeed	ft/sec
$V_t$dot	time rate of change in true airspeed	(ft/sec)/sec
V_{tf}	final true airspeed	ft/sec
V_{ti}	initial true airspeed	ft/sec
ν	viscosity	cinestokes
W	aircraft gross weight	lbs
Wf_s	standardized average normalized fuel flow	lbs/hr
W/δ	aircraft gross weight/ambient air pressure ratio	lbs
W_s	standard weight	lbs
δ_s	standard altitude ambient air pressure ratio	dimensionless
δ_t	test altitude ambient air pressure ratio	dimensionless
θ_s	standard ambient air temperature ratio	dimensionless
θ_t	test ambient air temperature ratio	dimensionless

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